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FINAL REPORT – PHASE II

PROPELLANT SELECTION FOR UNMANNED SPACECRAFT PROPULSION SYSTEMS

CONTRACT NASw-1644

VOLUME II

ANALYSIS OF PROPELLANT SENSITIVITY, SECONDARY PROPULSION, AND GROUND OPERATIONS

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
WASHINGTON, D.C.

Lockheed

MISSILES & SPACE COMPANY

A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION

SUNNYVALE, CALIFORNIA

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FOREWORD

This report was prepared by the Lockheed Missiles & Space Company, Sunnyvale, California, and contains the results of a study performed for the National Aeronautics and Space Administration, Office of Advanced Research and Technology, under Phase II of Contract NASw-1644, Propellant Selection for Unmanned Spacecraft Propulsion Systems. The report is printed in three volumes:

Volume I	Results, Conclusions, and Recommendations
Volume II	Analysis of Propellant Sensitivity, Secondary Propulsion, and Ground Operations
Volume III	Study of Propulsion Stage Commonality and Attitude Control Systems Requirements

CONTENTS

Section		Page
	FOREWORD	iii
	ILLUSTRATIONS	ix
	TABLES	xiii
	INTRODUCTION	1
1.0	PROPELLANT SENSITIVITY ANALYSIS	3
	1.1 Temperature Dependent K Factors	4
	1.2 Shadow Shields	16
	1.3 Engine Start Mode	26
	1.4 Thrust and Chamber Pressure Sensitivity	34
	1.5 Leak Rate Sensitivity	44
	1.6 Complexity Analysis	53
2.0	ANALYSIS OF SECONDARY VERSUS PRIMARY PROPULSION FOR MINOR ΔV REQUIREMENTS	61
	2.1 Analysis Approach	63
	2.2 Propulsion System Requirements	63
	2.3 Pressurization and Thermodynamic Effects on Primary Propulsion	65
	2.4 Propulsion Systems Descriptions	65
	2.4.1 Mode A, Throttled Main Engine	65
	2.4.1.1 Earth Storables – Mode A	65
	2.4.1.2 Sequence of Operations – Earth Storables – Mode A	68
	2.4.1.3 Space Storables – Mode A	70
	2.4.1.4 Sequence of Operations – Space Storables – Mode A	70
	2.4.1.5 Cryogens – Mode A	72
	2.4.1.6 Sequence of Operations – Cryogens Mode A	73

Section | Page

2.4.2 Primary Propulsion System – Single Burn – Modes B, C, and D ... 75

2.4.2.1 Single-Burn N_2O_4/A-50 Primary ... 76

2.4.2.2 Sequence of Operations, Single-Burn N_2O_4/A-50 Primary ... 78

2.4.2.3 Single-Burn Space Storables ... 80

2.4.2.4 Sequence of Operations – Single Burn Space Storables Primary ... 82

2.4.2.5 Single-Burn Cryogenic Primary ... 82

2.4.2.6 Sequence of Operations – Single Burn Cryogenic Primary ... 84

2.4.3 Secondary Systems Description – Mode B Common Propellant, Separate Tanks ... 87

2.4.3.1 N_2O_4/A-50 Secondary ... 87

2.4.3.2 Sequence of Operations – N_2O_4/A-50 Secondary ... 89

2.4.3.3 Space-Storable Secondary ... 92

2.4.3.4 Sequence of Operations – Space-Storable Secondary ... 94

2.4.3.5 Cryogenic Secondary ... 97

2.4.4 Propulsion Systems Descriptions – Mode C Common Propellant, Common Tanks ... 99

2.4.4.1 N_2O_4/A-50 Secondary ... 99

2.4.4.2 Sequence of Operations – N_2O_4/A-50 Secondary ... 101

2.4.4.3 FLOX/CH_4 Secondary ... 104

2.4.4.4 Sequence of Operations – FLOX/CH_4 Secondary ... 106

2.4.4.5 F_2/H_2 Secondary ... 106

2.4.5 Propulsion Systems Description – Mode D – N_2O_4/A-50 Secondary, Separate Tanks ... 106

2.4.6 Propulsion Mode E – Idle Mode Secondary ... 109

2.4.7 Propulsion Mode F – ACS Secondary ... 109

Section		Page
	2.5 System Performance	110
	2.6 Systems Complexity	114
	2.7 Performance and Complexity Summary	122
3.0	ANALYSIS OF GROUND OPERATIONAL REQUIREMENTS AND PROBLEMS	125
	3.1 Summary	127
	3.2 Ground Operations	127
	3.2.1 Launch Site Operations Flow	127
	3.2.2 On-Pad vs Off-Pad Propellant Loading	129
	3.2.3 Analysis of Ground Hold Heating and No-Vent	138
	3.2.4 FLOX Differential Boiloff Analysis	142
	3.2.5 Propellant/Propulsion Systems Operations Comparisons	142
	3.2.6 Propellant Hazards Comparison	150
	3.2.7 Propellant Spill Disposal	151
	3.2.8 Vent Gas Disposal	157
	3.3 Ground Facilities	158
	3.3.1 Basic ITL Capabilities	158
	3.3.1.1 Vertical Integration Building (VIB)	159
	3.3.1.2 Solid Motor Assembly Building (SMAB)	159
	3.3.1.3 Launch Complexes 40 and 41	162
	3.3.1.4 Launch Pad Structure	162
	3.3.1.5 Mobile Service Tower (MST)	162
	3.3.1.6 Umbilical Tower	162
	3.3.1.7 AGE Building	164
	3.3.1.8 Complex Support Building	164
	3.3.1.9 Vehicle Transporter	164
	3.3.1.10 Propellant Holding Area	165
	3.3.1.11 Gas Storage Area	165
	3.3.1.12 Air Conditioning	165
	3.3.2 ITL Modifications Recommended for Adding Centaur and New Propellants	165

Section		Page
	3.3.3 Basic Propellant Loading Systems	167
	3.3.4 Zero Vent Facilities	179
	3.3.5 Basic Design Considerations vs Propellants	179
	3.3.5.1 Propellant/Materials Compatibility	180
	3.3.5.2 Propellant Liquid Temperature Range	180
	3.3.5.3 Propellant Facilities Flexibility	180
	3.3.5.4 On-Pad Component Replacement	183
3.4	Operations and Facilities Complexity	186
3.5	References	188

ILLUSTRATIONS

Figure		Page
1	Mars Orbiter Cryogenic Stage Details	5
2	Mars Orbiter Space-Storable Stage Details	7
3	Thermal Conductivities for Multilayer Insulation	12
4	Thermal Conductivities Used in Computations	13
5	Aft Mounted Shadow Shield Configuration	19
6	Forward Mounted Radiation Shields	19
7	Forward Shadow Shield Structural Effects	20
8	Preliminary Shadow Shield Investigations	22
9	Heat Transfer as Function of Forward Shield Spacing	23
10	Shield and Tank External Temperature Distribution	25
11	Shadow Shield Heating Effects	28
12	Engine Start Mode Sensitivity Analysis	32
13	Thrust and P_c Sensitivity Summary	35
14	Engine Weight Variations With Thrust and P_c	38
15	F_2/H_2 Specific Impulse Variations With Thrust and P_c	39
16	$FLOX/CH_4$ Specific Impulse Variation With Thrust and P_c	40
17	N_2O_4/A-50 Specific Impulse Variation With Thrust and P_c	41
18	Thrust and P_c Sensitivity Results – N_2O_4/A-50 Pressure-Fed	45
19	Thrust and P_c Sensitivity Results – $FLOX/CH_4$ Pressure-Fed	46
20	Thrust and P_c Sensitivity Results – F_2/H_2 Pressure-Fed	47
21	Thermal Conductivity of Propellant Vapors	52
22	Mars Orbiter – Fluid Systems Schematic – Earth Storable Propellants	55
23	Mars Orbiter – Fluid Systems Schematic – Space Storable Propellants	56
24	Mars Orbiter – Fluid Systems Schematic – Space Storable Propellants (No Propellant Isolation)	57

Figure		Page
25	Mars Orbiter – Fluid Systems Schematic – Cryogenic Propellants	59
26	Mars Orbiter – Fluid Systems Schematic – Cryogenic Propellants (No F_2 Propellant Isolation)	60
27	Mars Orbiter Fluid System Schematic – Earth Storable Propellants – Single Main Engine Burn	77
28	Mars Orbiter Fluid Systems Schematic – Space Storable Propellants – Single Main Engine Burn	81
29	Mars Orbiter Fluid Systems Schematic – Cryogenic Propellants – Single Main Engine Burn	83
30	Mars Orbiter – Secondary Propulsion System Schematic – Separate Tankage, N_2O_4/A-50	88
31	Mars Orbiter – Secondary Propulsion System Schematic – Separate Tankage, FLOX/CH_4 or F_2/H_2	93
32	Mars Orbiter – Secondary Propulsion System Schematic – Integrated With Main Tanks, N_2O_4/A-50	100
33	Mars Orbiter – Secondary Propulsion System Schematic – Integrated With Main Tanks, FLOX/CH_4 or F_2/H_2	105
34	Secondary Propulsion Complexity and Weight Penalty	124
35	Integrate Transfer Launch Area, Plan View	126
36	Launch Site Operations – On-Pad Propellant Loading	128
37	Launch Site Operations – Remote Propellant Loading, N_2O_4/A-50	130
38	Launch Site Operations – Remote Propellant Loading, FLOX/CH_4, OF_2/CH_4, F_2/NH_3, OF_2/B_2H_6	131
39	Launch Site Operations – Remote Propellant Loading, F_2/H_2, O_2/H_2	132
40	Ground Hold Non-Vented System, LN_2 Spray Cooling Concept for NH_3, B_2H_6, CH_4, OF_2, O_2, FLOX, F_2	133
41	Ground Hold Non-Vented System, Cold He Heat Exchanger Concept for LH_2	134
42	Ground Hold Non-Vented System, Subcooled/Slush Concept for LH_2	137
43	Boiling Point Composition Program for FLOX Mixtures	143
44	Heat of Vaporization of Liquid-Fluorine – Liquid Oxygen Mixtures	143
45	FLOX Differential Boiloff Effects	144

Figure		Page
46a	(Part 1) Operations Flow Chart – N_2O_4/A-50, O_2/H_2, F_2/NH_3	145
46b	(Part 2) Operations Flow Chart – OF_2/B_2H_6, F_2/H_2, FLOX/ CH_4	147
47	Integrate Transfer Launch Area, Aerial View	160
48	Vertical Integration Building, Aerial View	161
49	Launch Complexes 40 and 41, Plan View	163
50	Titan IIID/Centaur Launch Complex	166
51	ITL Complex 40-41 Modified for Titan IIID/Centaur	171
52	N_2O_4 or A-50 Basic Propellant Loading System	170
53	O_2 Basic Propellant Loading System	173
54	H_2 Basic Propellant Loading System	174
55	CH_4 or NH_3 Basic Propellant Loading System	175
56	F_2 or OF_2 Basic Propellant Loading System	176
57	FLOX Basic Propellant Loading System	177
58	B_2H_6 Basic Propellant Loading System	178
59	Propellant Liquid Temperature Range	182

TABLES

Table		Page
1	Temperature Dependent Insulation Conductivity	11
2	Insulation Conductivity Performance Comparison	15
3	Shadow Shield Effects	17
4	Shadow Shield Weight Sensitivity	27
5	Engine Start Mode Effects	30
6	Engine Start Mode Sensitivity Analysis	33
7	Propulsion System Characteristics	37
8	Thrust and Chamber Pressure Sensitivity Analysis	43
9	Propulsion Modes	62
10	Propulsion Requirements	64
11	Primary System Weights for Single vs Multiburn	66
12	Weights of Secondary Propulsion System – N_2O_4/A-50 Secondary With Separate Tanks	90
13	Weights of Secondary Propulsion System – FLOX/CH_4 Primary and Secondary With Separate Tanks	95
14	Weights of Secondary Propulsion System – F_2/H_2 Primary and Secondary With Separate Tanks	98
15	Weights of Secondary Propulsion System – N_2O_4/A-50 Secondary With Common Tanks	102
16	Weights of Secondary Propulsion System – FLOX/CH_4 Secondary With Common Tanks	107
17	Weights of Secondary Propulsion System – F_2/H_2 Secondary With Common Tanks	108
18	Primary vs Secondary Propulsion for Minor ΔV – Stage Weight Changes From Mode A to Mode B	111
19	Primary vs Secondary Propulsion for Minor ΔV – Stage Weight Changes From Mode A to Mode C	112
20	Primary vs Secondary Propulsion for Minor ΔV – Stage Weight Changes From Mode A to Mode D	113

Table		Page
21	Summary of Critical Components and Flight Operations	117
22	Complexity Ranking	116
23	Propulsion Mode Analysis Results Summary	123
24	Configuration Data for Ground Hold Analysis	139
25	Ground Hold Analysis Results	140
26	Propulsion Operations Time Comparison	152
27	Personnel Hazards Comparison	153
28	Fire, Explosive, and Stability Hazards Comparison	154
29	Propellant Pad Storage Requirements	168
30	Modifications to Complex 40–41 for New Propellants	169
31	Propellant/Materials Compatibility	181
32	Propellant Facilities Flexibility	184
33	On-Pad Component Replacement Requirements With Propellants Drained	185
34	Complexity Comparison	187

INTRODUCTION

The Phase II study was divided into five major tasks plus reporting (Task V). Task I covered analysis of the sensitivity of propellants to various system perturbations. Task II entailed the comparison of using a secondary versus a primary propulsion system for minor ΔV requirements. In Task III the ground operational requirements and problems of the candidate propellants were studied. Task IV was the investigation of the feasibility of using a common stage, with minimum modification, for alternate space missions. Task VI was the identification of attitude control system requirements for the missions and configurations considered in Task IV. This volume presents the results of Task I, II, and III. Tasks IV and VI are discussed in Vol. III.

Section 1.0
PROPELLANT SENSITIVITY ANALYSIS

One of the factors affecting the choice of one propellant over other propellants is the sensitivity to various system perturbations. Several of these perturbations were investigated in Phase II. They include the sensitivity of insulation thermal conductivity to both inner and outer boundary temperatures, effects of shadow shields, sensitivity to engine start mode, effect of thrust and chamber pressure on pressure fed engine systems, sensitivity to propellant leakage, and the effect on complexity for each propellant combination.

In order to achieve some consistency in the analysis the Phase I Mars Orbiter, described in Lockheed Report K-19-68-6, was used as the baseline vehicle. This vehicle has an 8,000-lb-thrust propulsion system used to insert a spacecraft into an eccentric orbit about Mars. Examples of the Phase I Mars Orbiter stage design are presented in Figures 1 and 2 for cryogens and space storables respectively. Nominal parameters for the mission and stage are as follows:

- Payload 8,143 lb
- Mission duration 205 days
- ΔV total: 6,950 ft/sec

1st midcourse	164 ft/sec at 2 days
2nd midcourse	164 ft/sec at 165 days
Orbit insertion	6,294 ft/sec at 195 days
Orbit trim	328 ft/sec at 205 days

- Three-axis stabilization

The three propellant classes were represented by F_2/H_2, FLOX/CH_4 and N_2O_4/A-50 for all but the insulation conductivity analysis for which a broader spectrum of propellants was considered.

1.1 TEMPERATURE DEPENDENT K FACTORS

A study was conducted to evaluate the effect of using temperature dependent values of thermal conductivity for the tank insulation. Previous analyses were performed with constant values of conductivity selected on the basis of available test data. These conservatively high values assumed for conductivity were:

For H_2 tanks	2.5×10^{-5} Btu/ft-hr-°R
For F_2, O_2, OF_2, FLOX, and CH_4	5.0×10^{-5} Btu/ft-hr-°R
For earth storables	$10. \times 10^{-5}$ Btu/ft-hr-°R

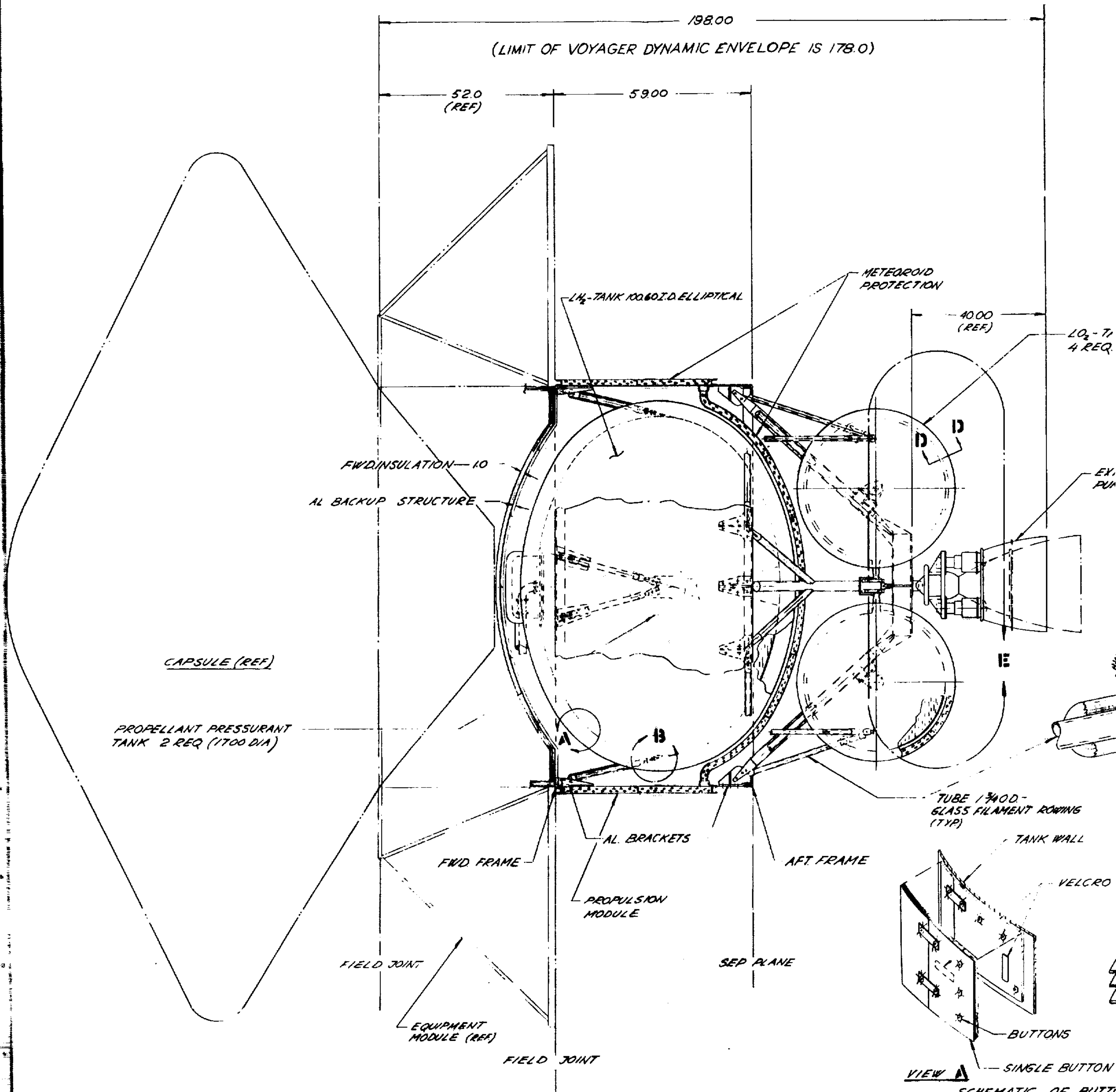
In comparing temperature-dependent values of thermal conductivity with the specific values used in the baseline case, the pump-fed Mars Orbiter vehicle was analyzed. Sun-on-tank orientation was utilized because of the greater sensitivity to K factors. F_2/H_2, O_2/H_2, FLOX/CH_4, OF_2/CH_4, and F_2/NH_3 were the propellant cases investigated.

1.1.1 Summary

Utilizing temperature-dependent insulation conductivity (K) factors in the analysis rather than a constant value for each propellant class resulted in K values changes by up to a factor of 2 or more. The fixed values were found to be conservative, with propulsion module weight reductions from the new K values as follows:

Propellant	Percent Weight Reduction
F_2/H_2	0.9
O_2/H_2	1.5
FLOX/CH_4	0.6
OF_2/B_2H_6	0.3
F_2/NH_3	0.4

This result tends to indicate the insenstivity of system weight to insulation conductivity changes.



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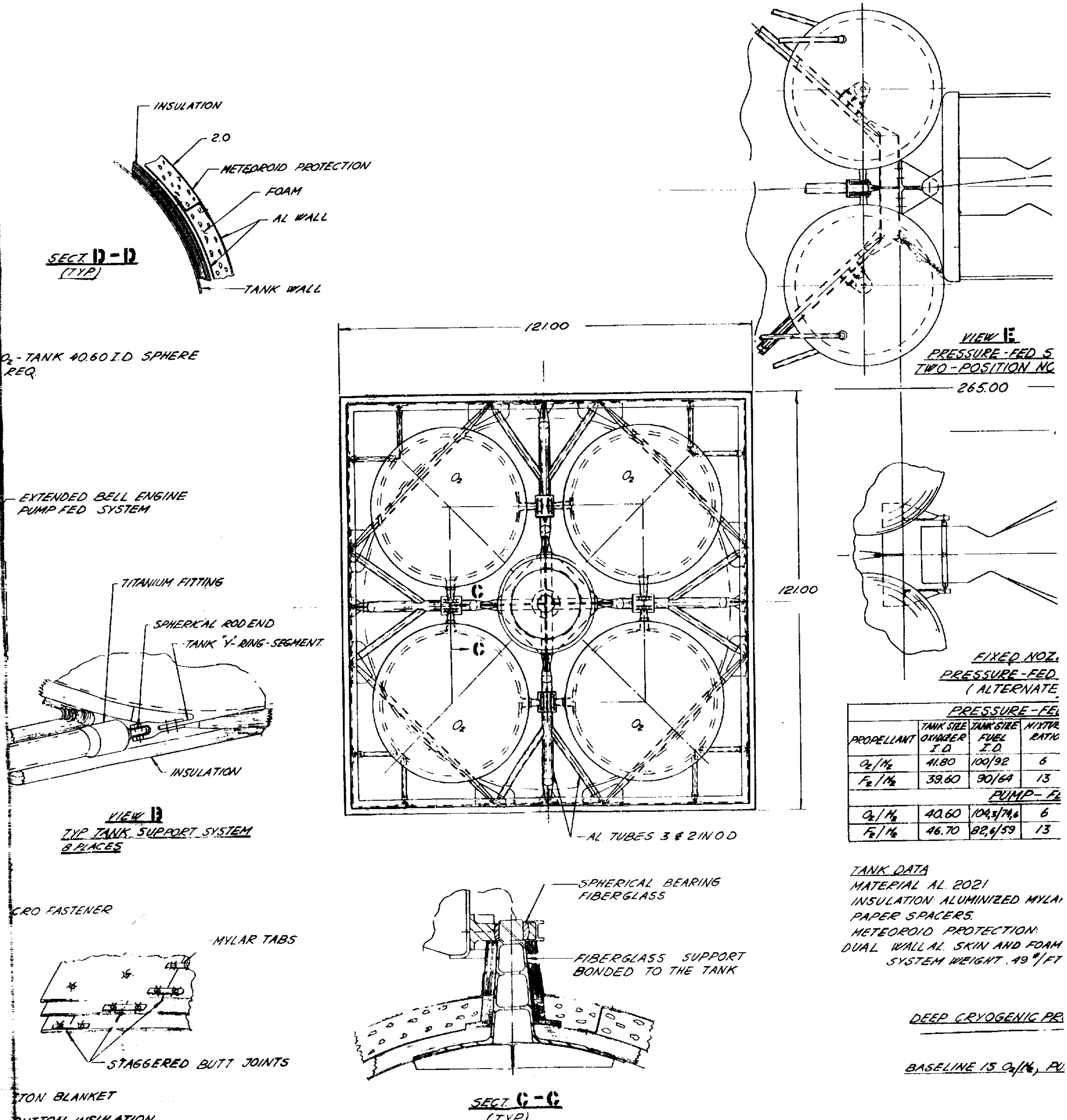

PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTUR RATIO
PRESSURE-FE			
O_2/H_2	41.80	100/92	6
F_2/H_2	39.60	90/64	13
PUMP-F			
O_2/H_2	40.60	104.3/74.6	6
F_2/H_2	46.70	82.6/59	13

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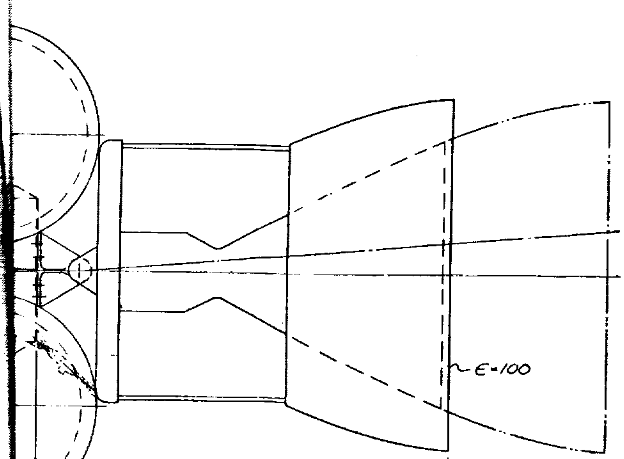
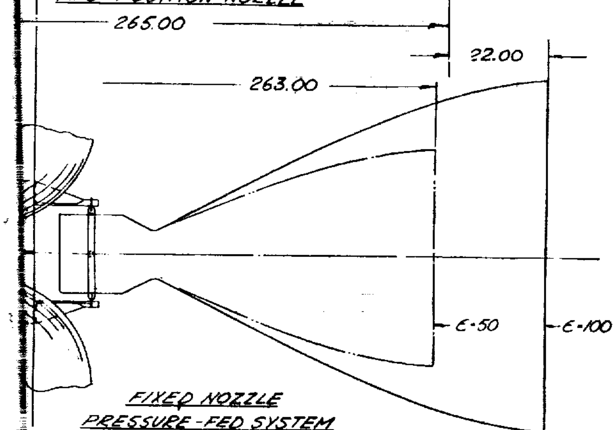
Fig. 1 Mars Orbite



VIEW E
PRESSURE-FED SYSTEM
TWO-POSITION NOZZLE



FIXED NOZZLE
PRESSURE-FED SYSTEM
(ALTERNATE)

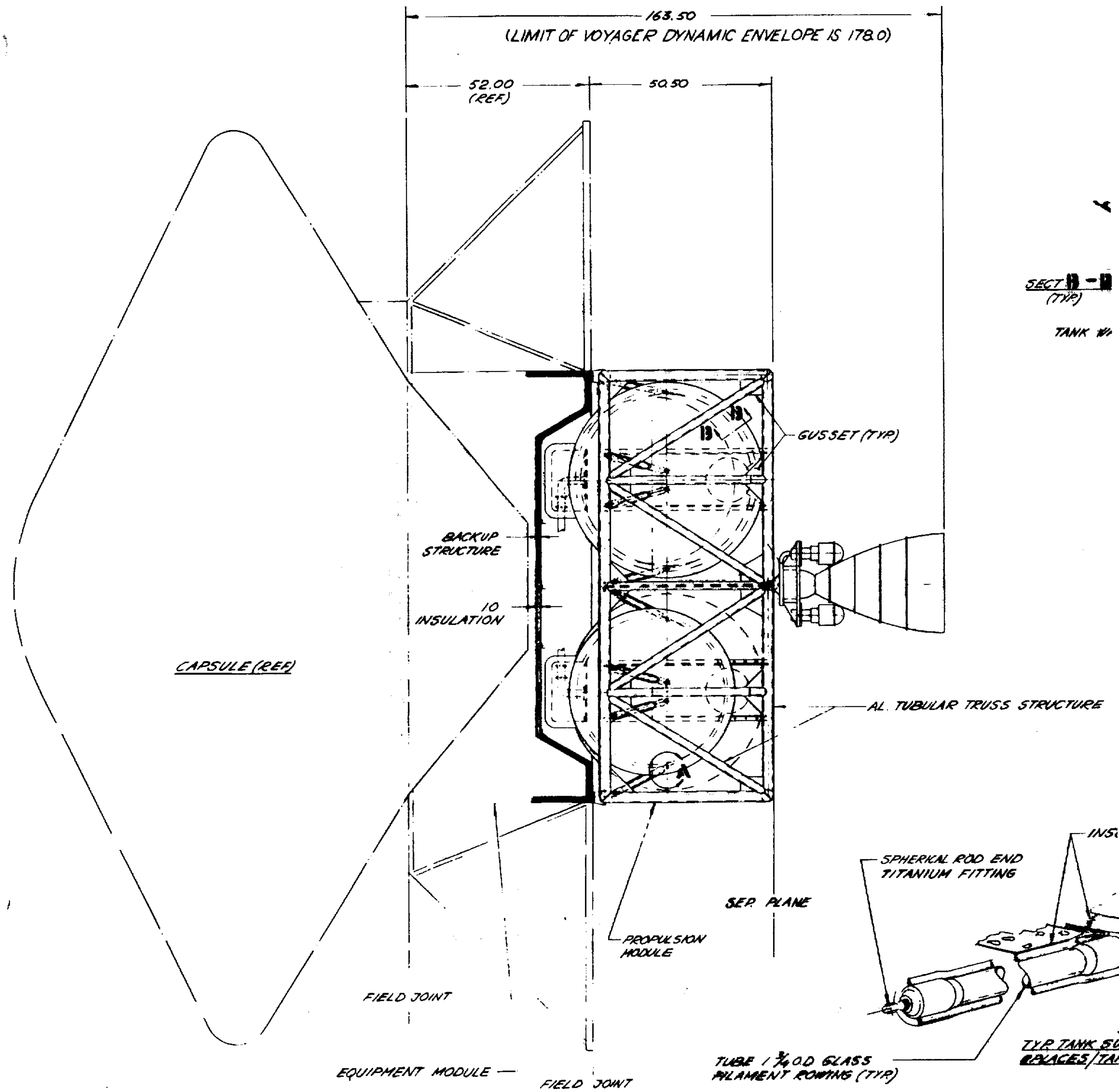
NT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO	VEHICLE LENGTH IN.
PRESSURE-FED SYSTEM							
	41.80	100/92	6	445	100	100	265
	39.60	90/64	13	442	100	100	245
PUMP-FED SYSTEM							
	40.60	104.3/74.6	6	451	900	100	198
	46.70	82.6/59	13	468	900	100	187

ATA
AL AL. 2021
ION: ALUMINIZED MYLAR AND DEXIGLAS
SPACERS.
OID PROTECTION:
LL AL. SKIN AND FOAM BETWEEN
STEM WEIGHT .49 #/FT²

DEEP CRYOGENIC PROPELLANTS

ASELINE IS O_2/H_2, PUMP FED.

1 Mars Orbiter Cryogenic Stage Details



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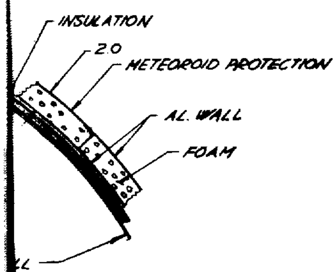
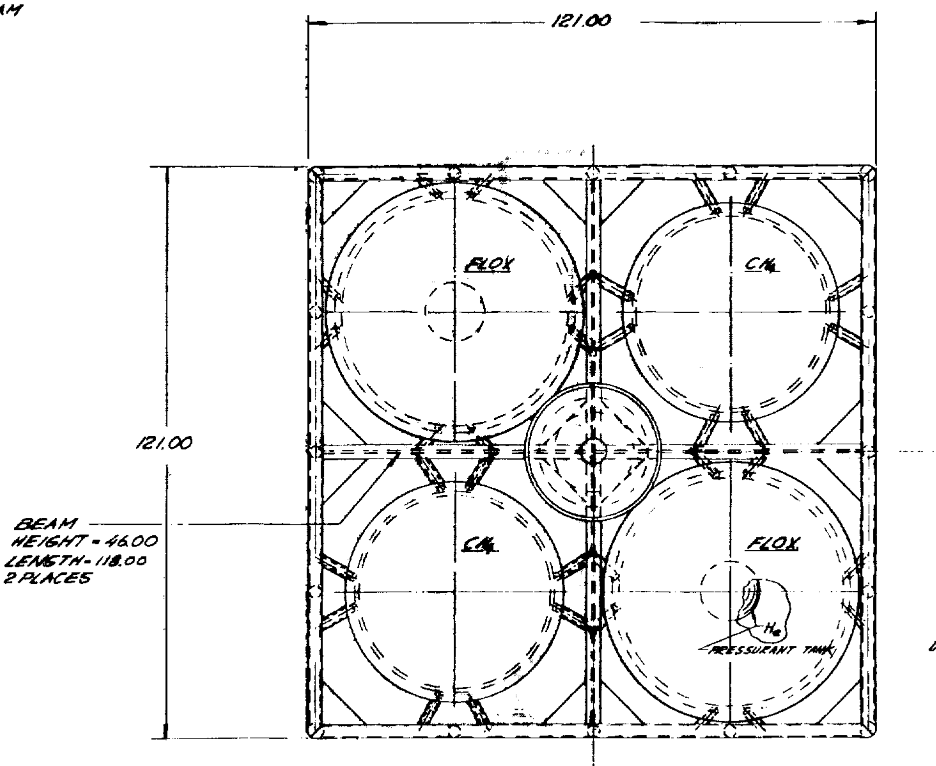
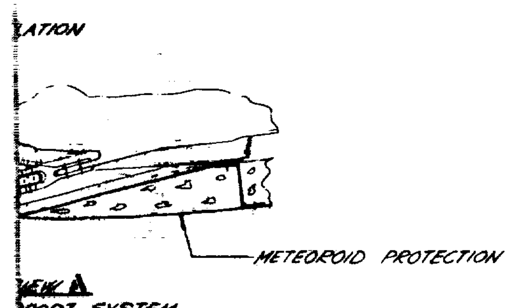




TANK DATA
MATERIAL AL 2021
INSULATION: ALUMINIZED MYLAR AND DEXIGLAS PAPER SPACERS.
METEOROID PROTECTION:
DUAL-WALL AL SKIN AND FOAM BETWEEN
SYSTEM WEIGHT .46 #/FT²
ENGINE: PUMP-FED SYSTEM



PROPELLANT	TANK SIZE OXIDIZER I.D.	TANK SIZE FUEL I.D.	MIXTURE RATIO	I_{sp}	ENGINE DATA CHAMBER PRESSURE	ENGINE DATA EXPANSION RATIO
OF_2/CH_4	46.80	41.60	5.3/1	410	600	100
F_2/NH_3	46.20	40.30	3.3/1	408	1500	100
$FLOX/CH_4$	47.80	40.80	5.75/1	410	600	100

SPACE STORABLE PROPELLANTS

BASELINE IS $FLOX/CH_4$ SYSTEM, PUMP FED

Fig. 2 Mars Orbiter Space-Storable Pump-Fed Stage Details

FOLDOUT FRAME 2

1.1.2 Thermodynamic Analysis

Equations were recently developed for multilayer insulation conductivity as a function of both inner and outer boundary temperatures under NASA Contract NAS 8-20758, "Investigations Regarding Development of a High Performance Insulation System." Several equations were developed from flat-plate calorimeter conductivity values for several different types of multilayer insulation systems.

The baseline heating rates obtained with the conductivity values used in Phase I provided a relatively accurate assessment of the system performance, but did not point up small differences between individual space storable systems which result from the variation in insulation conductivitiy with boundary temperature changes. Accurately computed conductivity values as a function of both inner and outer boundary temperatures were used in a refined analysis so the relative performance of the space storable systems could be obtained.

The insulation system assumed for this refined analysis (closely comparable to the baseline used in Phase I) consists of double-aluminized Mylar radiation shields with dexiglas spacers at a layer density of 80/inch. Subsequent studies of the Commonality Stage, reported in Volume III, were based on the use of a lighter weight insulation consisting of double-aluminized mylar with tissuglas spacers. This insulation weighs 2.3 lb/ft^3 as compared with the 4.5 lb/ft^3 system of Phase I.

The equation for effective conductivity of this insulation was:

$$K_{eff} = \left[4.58 \times 10^{-2}(80)^2(T_m) + \frac{2.7 \times 0.173 \times 10^{-8}\left(T_h^2 + T_c^2\right)(T_h + T_c)t}{(N - 1)(2/\epsilon - 1)}\right]$$

$$= \left[293.12\ T_m + \frac{4.671 \times 10^{-9}\left(T_h^2 + T_c^2\right)(T_h + T_c)\,t}{(N - 1)(2/\epsilon\ 1)}\right] \quad \text{Btu/hr-ft-°R}$$

where

T_m = $(T_h + T_c)/2$

N = total number of layers

t = thickness (ft)

ϵ = 0.036

T_h = hot side temperature, °R

T_c = cold side temperature, °R

This equation was derived from flat plate calorimeter data. In the analysis the conductivity computed with this equation was multiplied by a factor of 2.2 to account for degradation. The factor of 2.2 results in an effective thermal conductivity of 2.5×10^{-5} Btu/hr-ft-°R for applicable H_2 temperatures.

The equation was programmed into the thermal network analyzer program used to solve for heat flow and temperatures. Insulation temperatures were computed using this conductivity equation in an iterative energy balance. The insulated surface is divided into several nodes for accurate analysis, but only the area-weighted average outer surface temperature is reported in Table 1 and Figs. 3 and 4. The area weighted average temperature for a tank is the sum of the products of each node area multiplied by its temperature, all divided by the total tank area. The inner insulation temperature is assumed to be the liquid propellant temperature.

Figure 3 shows conductivity as a function of cold side (inner) temperature with warm side (outer) temperature as a parameter. The variation in conductivity with thickness is on the order of 1 to 2 percent for a range of insulation thickness of 1 to 4 in. and, therefore, is not included as a parameter. Figure 4 shows the thermal conductivity values previously assumed and the values computed in the refined analysis. The conductivity values computed for all but the earth storables give average values less than the constant conductivity values used in the baseline analysis. This further

Table 1

TEMPERATURE DEPENDENT INSULATION CONDUCTIVITY

	Insulation Surface Temperature		Conductivity Btu/Hr-Ft-°R × 10^{-5}			Change in Weight*	
	Earth (°R)	Mars (°R)	Baseline	K(T) Earth	K(T) Mars	Residuals (lb)	Inert (lb)
F_2	302	252	5.0	2.54	2.03	- 5.2	- 3.7
H_2	389	330	2.5	2.66	1.98	- 8.0	-11.5
O_2	302	252	5.0	2.54	2.03	- 6.8	+ 0.2
H_2	389	330	2.5	2.66	1.98	+ 5.0	-45.
FLOX	384	326	5.0	3.58	2.80	- 9.8	- 0.5
CH_4	384	326	5.0	4.08	3.23	- 1.5	- 1.0
OF_2	384	326	5.0	4.44	3.56	- 3.4	+ 0.4
CH_4	384	326	5.0	4.08	3.23	- 4.1	- 0.5
F_2	384	326	5.0	3.56	2.77	- 0.8	- 7.0
NH_3	555	465	10.0	13.2	10.2	0	0

*Relative to baseline

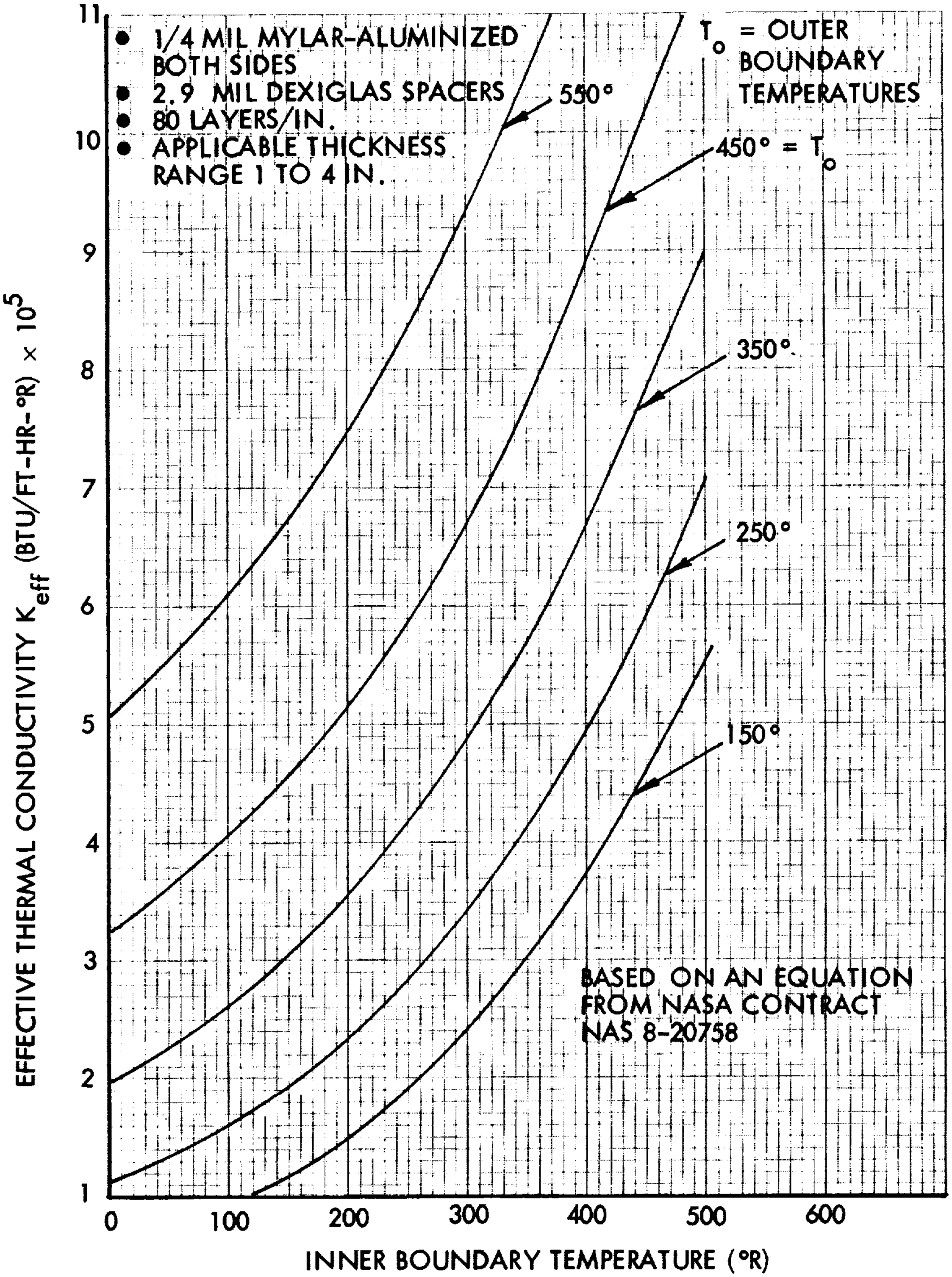


Fig. 3 Thermal Conductivities for Multilayer Insulation as a Function of Boundary Temperatures

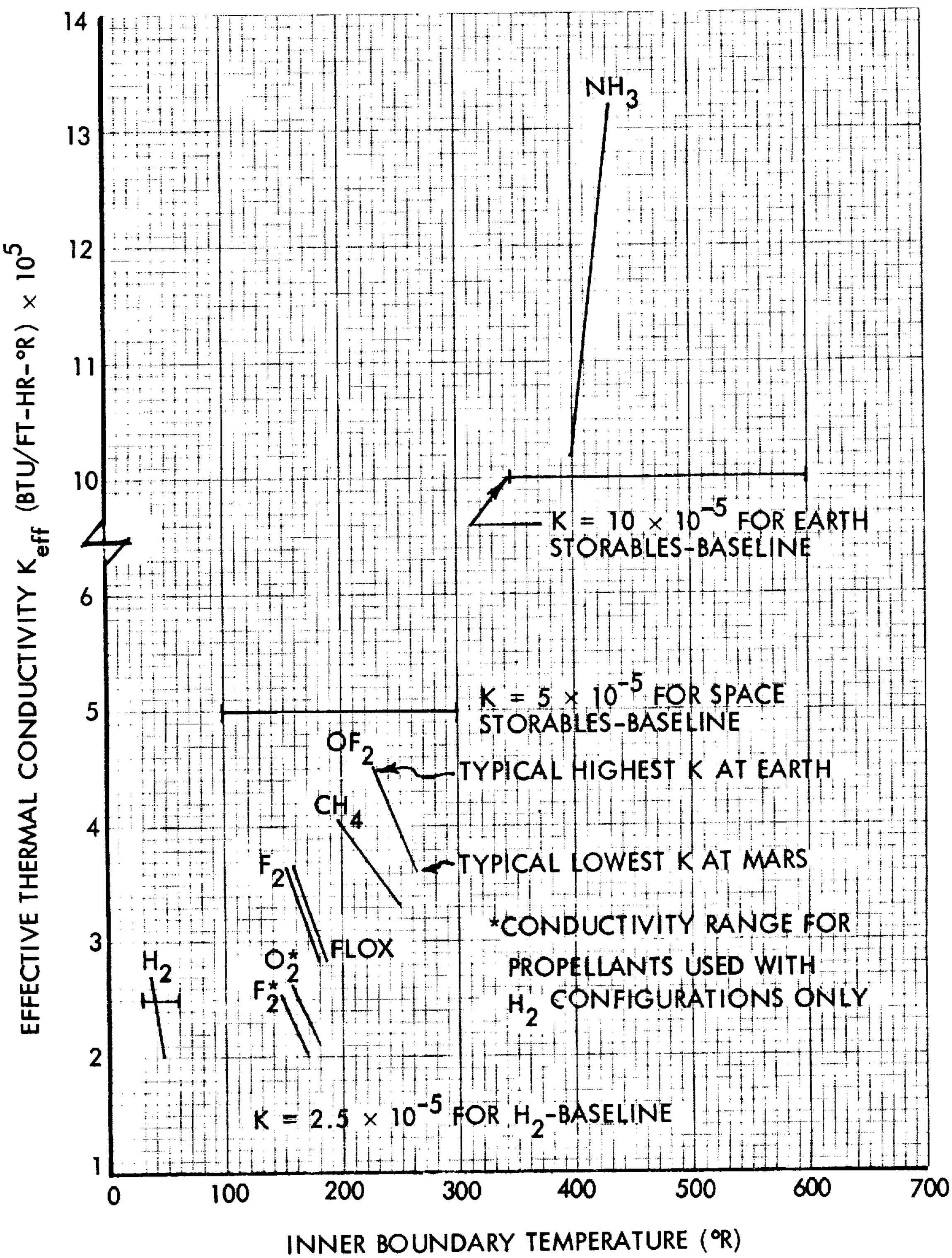


Fig. 4 Thermal Conductivities Used in Computations

indicates the conservatism of the Phase I analysis. The refined conductivity values decrease with distance from the sun, cuasing the greatest propellant heating near Earth and less heating near Mars. The figure shows that the oxidizer tanks in the O_2/H_2 and F_2/H_2 configurations have conductivities nearly as low as the H_2 tanks. This result is due to the external surface temperature of the oxidizer tanks being lower than both the H_2 tanks and the oxidizer tanks of the other configurations. In the hydrogen systems the oxidizer tanks are relatively well isolated from the payload and solar arrays and have a good view to space compared to the other configuration.

The average surface temperatures listed on Table 1 at Earth and at Mars are the integrated average of all the surface node temperatures. The external temperatures decrease substantially more than the internal or propellant temperature increases with time, therefore, the average insulation temperature decreases with time. The net result is a decrease in the insulation conductivity with time.

The difference in thermodynamic system weights for the constant conductivity baseline and the refined temperature-dependent conductivity analyses are listed on the table. The differences in weights for these two analyses are slight although, in general, there is a reduction in the weight with temperature-dependent conductivity. This trend indicates a degree of conservatism in the first analyses and shows very little net effect on the ratings of the space storable propellants. Performance of the hydrogen system tends to be slightly better than before in this particular comparison.

1.1.3 Vehicle Analysis

The impact of a refined thermal conductivity analysis is manifest in new lower vehicle weights. Although the changes are very small, there are some slight changes in insulation thickness and also somewhat more significant changes in tank pressure, thereby reducing the vapor pressure for the refined analysis. Table 2 shows a comparison of the vehicle weights for the baseline and the temperature-dependent K factors. The maximum difference obtained was 125 lb for the O_2/H_2 propulsion module. All other weight differences were significantly less.

Table 2

INSULATION CONDUCTIVITY PERFORMANCE COMPARISON
Mars Orbiter Vehicle, Pump-Fed, Nonvented, Sun-on-Tank

Propellant	Baseline			Temperature Dependent K Factor			Percent Weight Change
	Tank Max. Operating Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Tank Max. Operating Pressure (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	
F_2 H_2	40 130	1-1/8 4-5/8	7238	78 174	1 4	7173	-0.9
O_2 H_2	58 96	3/4 4-5/8	8477	42 100	1 4-1/4	8352	-1.5
FLOX CH_4	57 107	1-1/2 3/4	7968	56 81	1-1/2 1	7918	-0.6
OF_2 CH_4	45 107	1-1/8 3/4	7874	58 79	1 1	7847	-0.3
F_2 NH_3	59 16	1-3/8 Min.	7993	55 16	1-1/2 Min.	7964	-0.4

Min. = 1/2-in. foam

1.2 SHADOW SHIELDS

An analysis was conducted in which two basic shadow shield concepts were developed in order to determine possible performance gains by reducing propellant heating to levels lower than those obtained with the baseline designs. The pump-fed, Mars Orbiter vehicle with F_2/H_2 propellants was utilized. Only F_2/H_2 was considered because it represents the critical case. The two basic shadow shield concepts studied were an aft shield with either Optical Solar Reflector (OSR)(α/ϵ = .05/.80) or white paint (α/ϵ = .3/.95) for the sun-on-tank configuration, and a forward shield for the sun-on-capsule orientation in which the spacing and number of shields between the payload and propulsion module was varied.

Designs for these concepts were developed, a thermodynamic analysis conducted, and a performance evaluation made in order to compare the shadow shield concepts with the baseline system.

1.2.1 Summary

A synthesis of the structural parameters and the thermodynamic characteristics was developed in order to define the effects of shadow shields on propulsion module design and weight. Scaling laws were developed for the design and a performance analysis conducted. The resulting tank operating pressures, insulation thicknesses, and propulsion module weights are compared with the baseline in Table 3. The comparison indicates that the two shadow shield configurations are approximately equal in performance and yield a 3 percent weight saving over the baseline case. Although the forward shield provides better thermal isolation than the aft shield it sustains a greater structural penalty and consequently yields almost the same final result. The significant aspect of this analysis, however, is that even for a hydrogen system with unvented tanks on a trip to Mars, shielding can be provided advantageously so that the insulation thickness, and therefore the mission heat input, is reduced to such low levels that insulation requirements are actually dictated by practical considerations such as prelaunch control.

Table 3

SHADOW SHIELD EFFECTS

F_2/H_2 PROPELLANT

CONFIGURATION	TANK OPERATING PRESSURE (PSIA)		INSULATION THICKNESS (IN.)		PROPULSION MODULE	
	F_2	H_2	F_2	H_2	WEIGHT (LB)	Δ
BASELINE (NO SHIELD, SUN ON TANK)	52	60	1-1/4	3	6,990	(0)
AFT SHIELD (SUN ON TANK)	60	61	3/4	1	6,760	(-230)
FORWARD SHIELD (SUN ON CAPSULE)	<15	27	1/2	1	6,764	(-226)

1.2.2 Design

The design concept for the aft-mounted shield with the propulsion system oriented toward the sun is shown in Fig. 5. The aft-mounted shield was assumed to consist of multilayer insulation, 1/2-in. thick, supported in place on a structural frame. The weight of the insulation is 15 lb, and the support frame 5 lb. Consequently a 20-lb penalty was assigned for this design concept.

The design concept for the forward mounted shield located between the spacecraft capsule and the propulsion module is shown in Fig. 6. The selected design consists of three radiation shields spaced 3 in. apart and thereby adding one foot to the overall spacecraft system length.

Two structural concepts were developed to obtain the space separation required for the forward shield. One concept consists of a rigid fiberglass truss which yields very low heat leaks, but increases in weight almost linearly with separation distance. The other, a more interesting concept, consists of fiberglass telescoping struts mounted in aluminum tubes. In this concept the separation distance is zero during launch and the payload is cranked out in space. This design has a relatively large fixed weight because of the drive mechanism but increases only slightly with separation distance. The weight as a function of distance for both of these concepts is shown in Fig. 7.

1.2.3 Thermodynamic Analysis

Shadow shields can be incorporated into a vehicle design to introduce a large thermal resistance (radiation) between energy sources and the propellant tanks. All propellants studied for this vehicle concept except for hydrogen can be kept at or near their liquid equilibrium temperature level through spacecraft orientation alone; therefore, there is little to be gained by using shadow shields on any except hydrogen systems. For this reason the shadow shield analysis was conducted for the F_2/H_2 systems only.

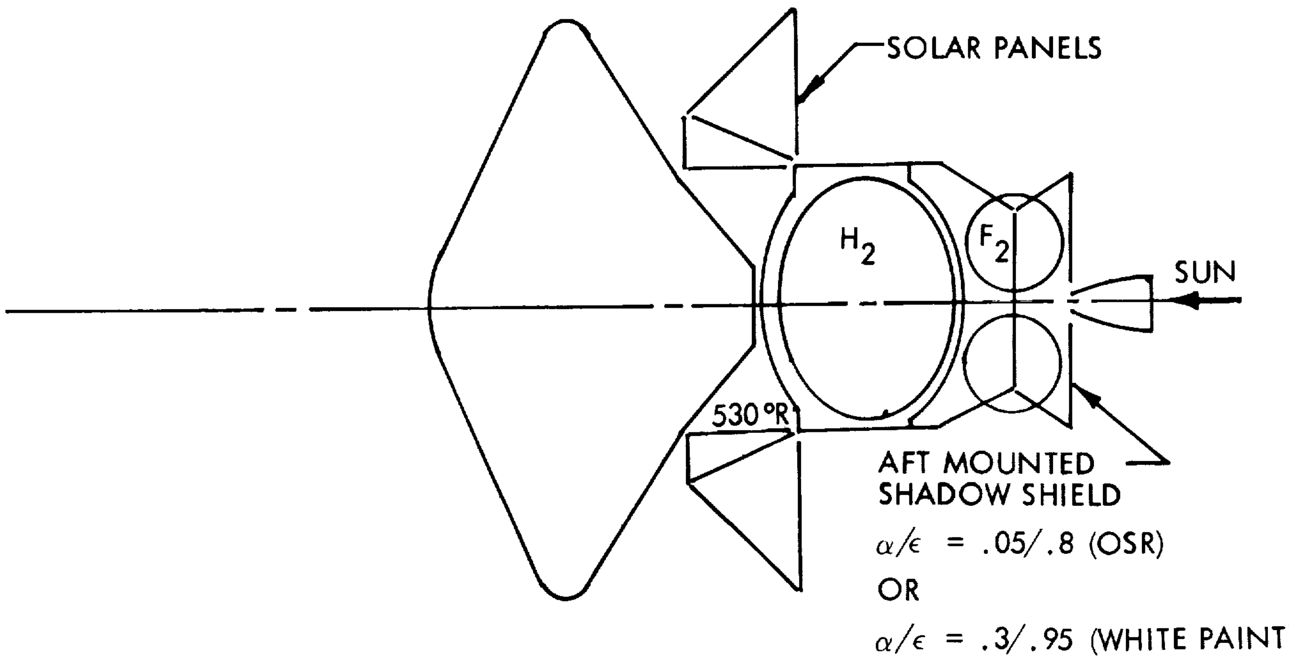


Fig. 5 Aft Mounted Shadow Shield Configuration

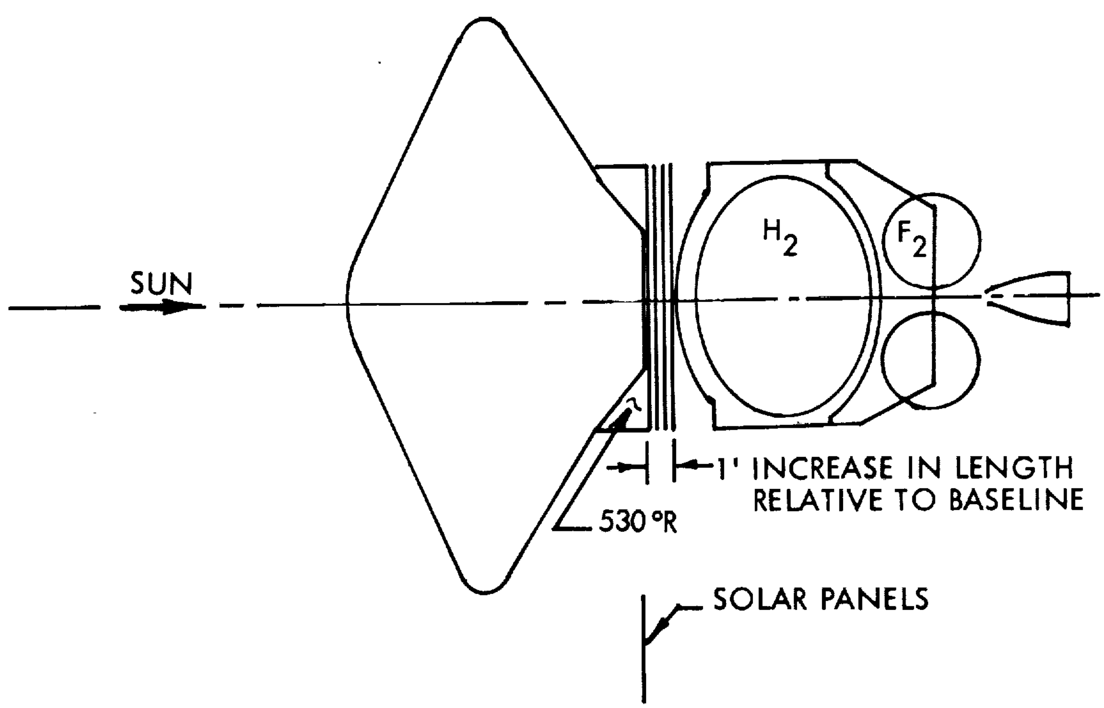


Fig. 6 Forward Mounted Radiation Shields

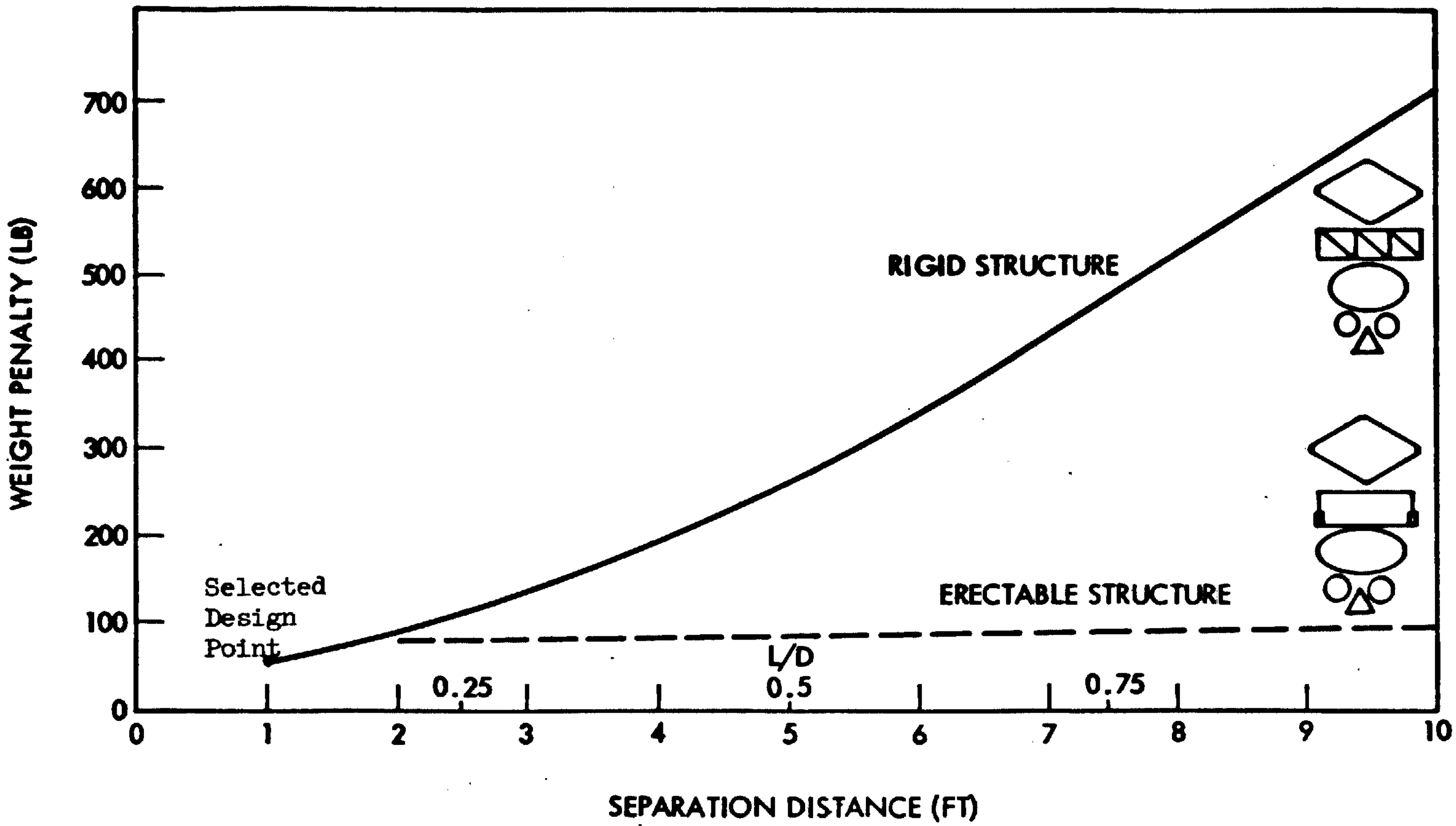


Fig. 7 Forward Shadow Shield Structural Effects

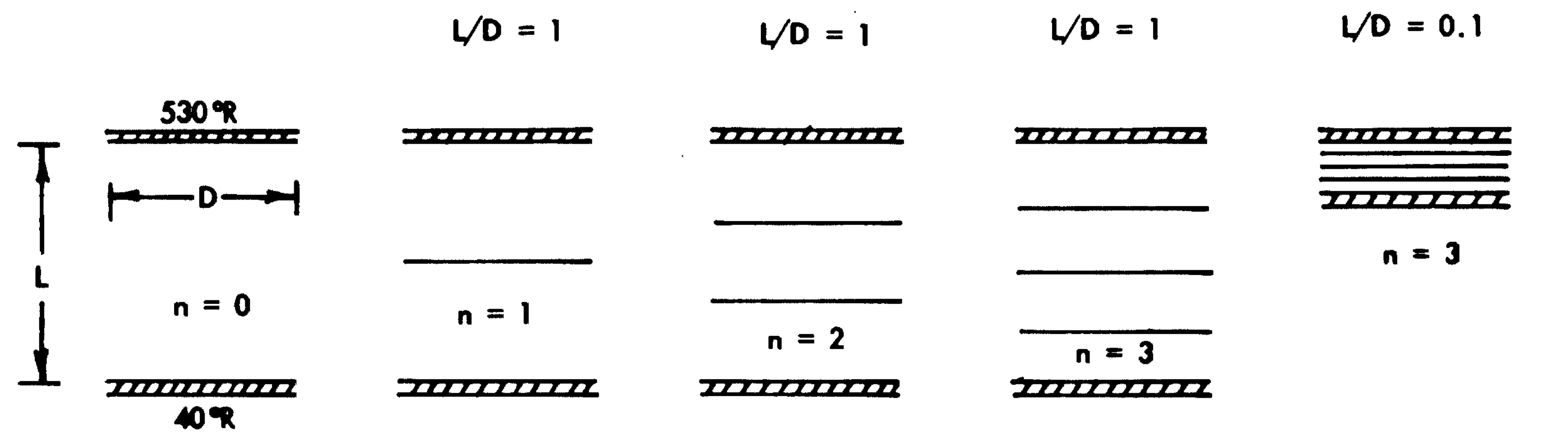
The geometry and spacing of the forward mounted shield was selected on the basis of a preliminary analysis conducted to determine how many shields should be used and what the spacing should be. Cases considered in the preliminary analyses are illustrated in Fig. 8. Figure 9 shows heat flux as a function of L/D, plus heat flux for L/D = 0.1 and three radiation shields. Spacings ranging from an L/D = 0.1 to 1.0 were investigated. It was determined that a spacing with an L/D of 0.1 or greater containing very few shields (3 or more) can essentially eliminate radiation heat transfer to the propellant tanks. With an L/D of 1.0 and low emittance surfaces (ϵ = 0.05) one shield located within this space can reduce the net radiation heat transfer to zero. More shields can actually extract energy. This study is idealized of course since structural penetrations and solar panels are not included. For the selected configuration, the actual vehicle characteristics were included, however.

Heat flux and shield temperatures were computed in the preliminary analyses with simplified models using the thermal analyzer program. When shields were spaced close together each was described by three (3) equal area concentric nodes.

Selection of the final forward shield configuration, three shields spaced equally in a one foot extension, was based on both practical and performance considerations. The number of shields was not optimized, and one or two shields rather than three would result in only a slight performance degradation.

From previous analyses it was estimated that by eliminating heat transfer to the H_2 a maximum reduction in thermally related weights of about 200 pounds could be achieved. The structural weight penalty for the forward mounted shield was shown in Fig. 7. Using the rigid fiberglass truss, a one foot space results in an inert weight of about 59 lb. The relatively large reduction in heat transfer, modest inert weight increase, and simplicity for a one foot spacing led to its selection.

With the aft shield concept, where the engine is exposed to the sun, there is also only slight heating of the propellants. In this case 1.0 in. of insulation was selected for the H_2 tank because of practical aspects, such as, pre-launch requirements. Previous



L/D	Q_{Net} (Btu/hr-ft^2)
.1	$.26 \times 10^{-2}$
.5	.55
.25	.97
.10	1.62

$Q_{Net} = 0$

$Q_{Net} \leq 0$

$Q_{Net} \leq 0$

$Q_{Net} = 1.4 \times 10^{-2}$ Btu/hr-ft^2

NOTES:
1. Q_{Net} FOR BASELINE DESIGN WITH SUN ON CAPSULE IS 31.7×10^{-2} Btu/hr-ft^2
2. n = NUMBER OF INTERMEDIATE RADIATION SHIELDS
3. EMISSIVITY OF ALL SURFACES AND SHIELDS = 0.05

Fig. 8 Preliminary Shadow Shield Investigations

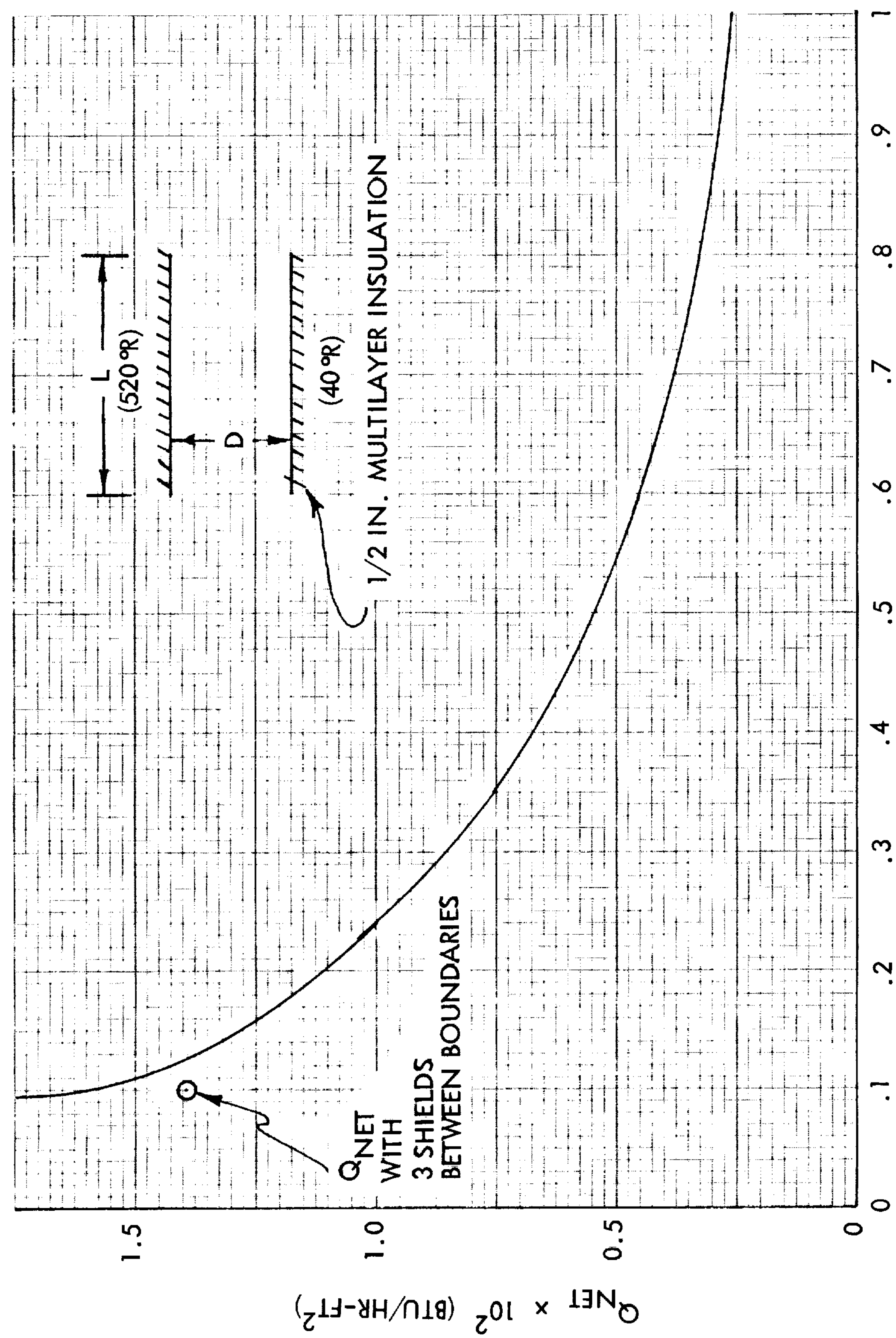


Fig. 9 Heat Transfer as Function of Forward Shield Spacing

analyses have shown that about 1.0 in. of insulation, purged with helium, limits prelaunch heating rates to manageable levels for topping or recirculating propellant. The F_2 tank surface temperatures can be held at or near the liquid equilibrium temperature with either an OSR or white paint shield surface. An insulation thickness of 1/2 in. was selected as a practical limit on the basis of prelaunch heating as near optimum for the F_2 tanks.

The aft mounted shield was assumed to consist of multilayer insulation (1/2 in. thick) supported in place on a structural frame. Weight of the insulation, based on a density of 4 lb/ft^3, is 15 lb. An alternate and lighter weight system could consist of three (3) radiation shields supported in place by some technique which prevents or minimizes contact of the shields. This would provide a thermal resistance equivalent to that of the 1/2-in. thick multilayer insulation. The aft facing side was assumed capable of supporting an OSR surface or it could be painted. The effects of heating during engine burn were not evaluated.

Detailed analyses of the selected forward and aft mounted shield configurations were conducted using thermal models and the thermal analyzer program to compute external heating rates and surface temperatures. The thermal models were modifications of those developed during previous studies. Incorporation of the shields required complete rework of the radiation networks and significant changes in the conduction network of the forward shield model where structural changes occurred. In the forward shield concept each shield was represented by three (3) equal area concentric nodes, Fig. 10. Significant temperature gradients from the center to the outer radius of the shields exist when the shields are relatively close together. The structural supports (fiberglass tubes) are not included in the radiation network. This results in computation of conservatively high penetration heat leaks because energy would actually be radiated away from the struts which view space. By accurately accounting for radiation losses from the struts in the forward shield configuration the conduction heat leak might be reduced by 50 to 70 percent. In the analysis presented here, the structural heat leaks are so small anyway, that accounting for this possible additional reduction is of little significance.

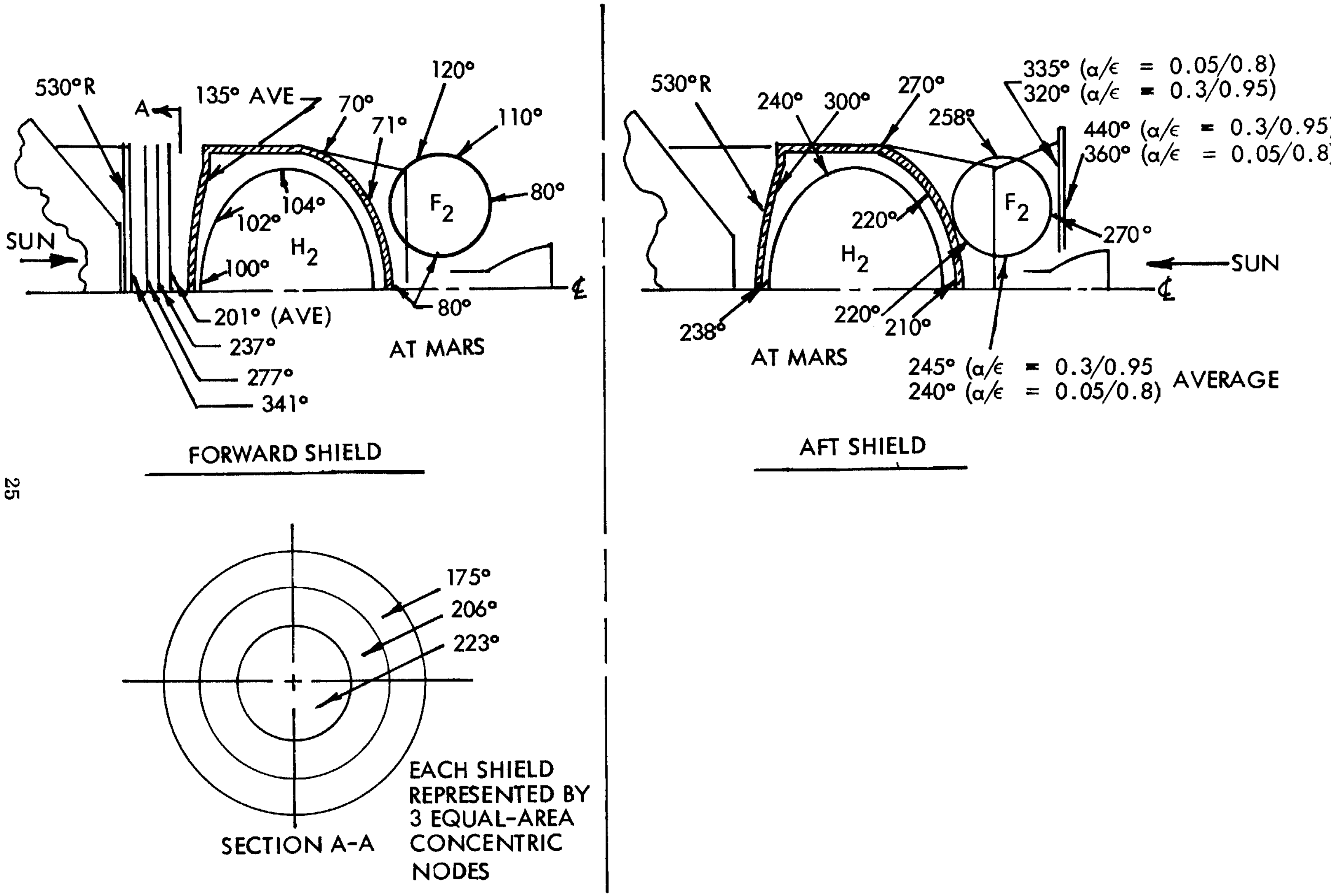


Fig. 10 Shield and Tank External Temperature Distributions

The thermal-pressurization program was used to compute pressures, insulation thicknesses and thermally related system weights. Table 4 presents this data and includes the weight values for the parameters considered for both shield configurations. With the forward shield there is very little hydrogen heating and the F_2 is actually cooled. An insulation thickness of 1.0 in. for the H_2 and 0.5 in. for the F_2 were selected as minimum practical values for prelaunch control, to minimize heating during maneuvers, and to prevent F_2 freezing.

It is apparent that use of the sun shields located either forward or aft, are an effective way of reducing propellant heating. This in turn results in requirements for very nominal, 1 in. or less, insulation thicknesses and relatively low tank pressures. Figure 11 compares the baseline H_2 tank system weight with the aft and forward shield systems. The absolute minimum weight system with 1 in. of insulation is also shown for reference. It can be seen that the absolute minimum weight system is closely approached by use of shields and that reduction of heat transfer by all shield concepts considered is so effective that differences between concepts are very small.

All analyses conducted to date have been based on the assumption that the spacecraft is sun oriented during the Mars orbit phase. Because of the relatively high orbit (1,000 × 20,000 km) the effect of Mars emission on the propellant tanks has been assumed negligible. This assumption is quite valid with a sun-on-tank configuration. However, it is less valid for sun-on-capsule orientation or where shadow shields are employed because the tank surface temperatures are so low that Mars emission can become significant. Little error is introduced for the sun shielded cases of this study because of the short duration of the Mars orbit and the insulation thicknesses selected. It should be noted that neglecting Mars emission heating is not conservative and in future studies, particularly for long orbit stay periods, the effect of Mars emission should be included.

1.3 ENGINE START MODE

In the Phase I study a tank-head-idle start mode was assumed in which any combination of liquid and vapor in the tank can be utilized in an as-is condition by the engine to

Table 4

SHADOW SHIELD WEIGHT SENSITIVITY

Configuration		Weight, lb					Area Weighted Tank Surface Temp. °R	
		Vapor	Pressurant	Insulation	Tank	Σ	Earth	Mars
Baseline (No shield, sun on tank)	F_2	21	6.4	25.9	77.3	130.6	302	252
	H_2	35.3	–	174.8	145	355.1	389	330
Aft Shield (Sun on Tank)	F_2	20.8	2.4	15.5	77	115.7	290	240
	H_2	27.9	–	53	137	217.9	275	231
Forward Shield (Sun on Capsule)	F_2	1.6	2.1	10.2	76.8	90.7	<150	<150
	H_2	14.8	–	53	137	204.8	94	89

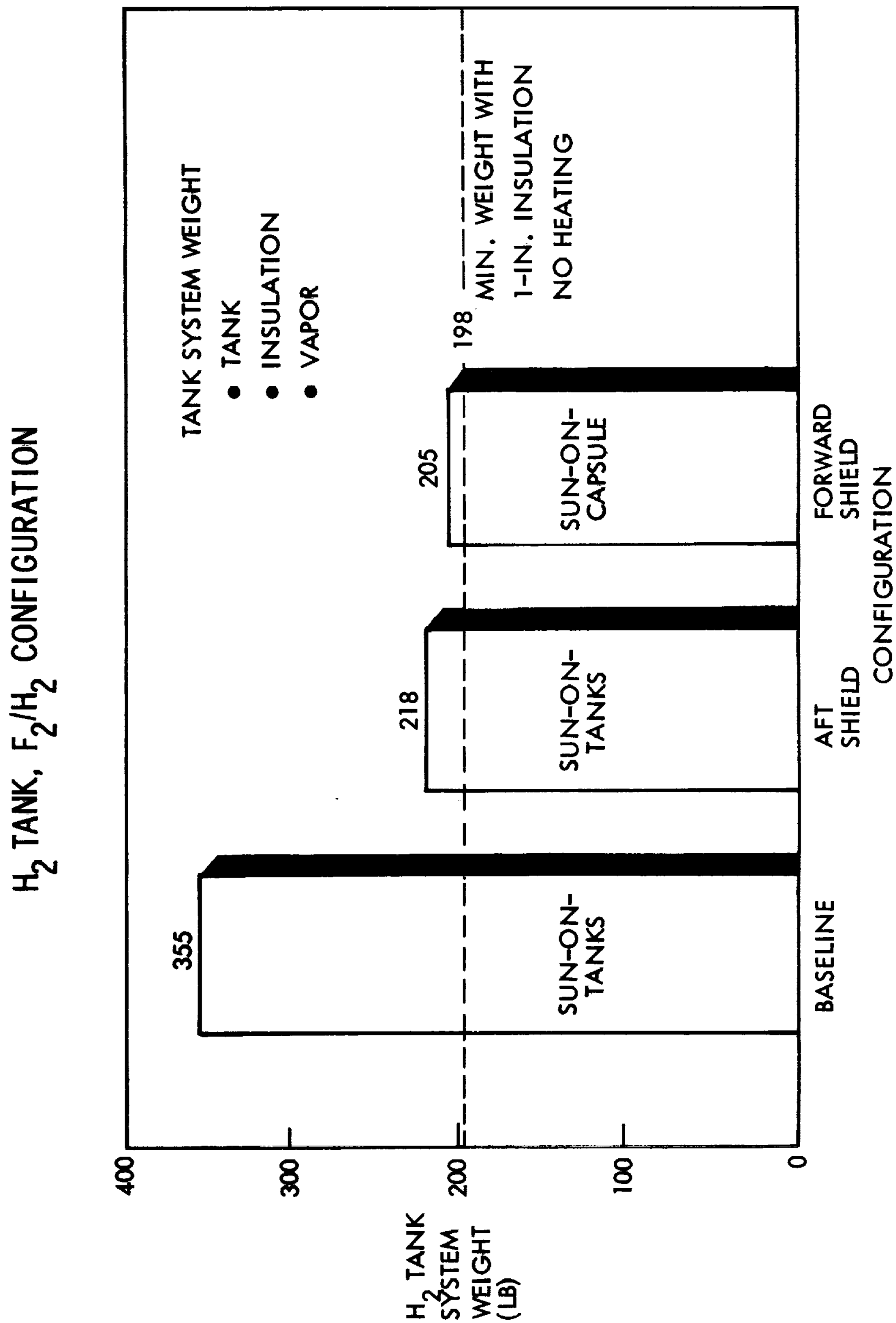


Fig. 11 Shadow Shield Effects

initiate operation. This method of operation is not always possible and penalties for alternative methods of operation were investigated. This was done for the Mars Orbiter pressure-fed system with F_2/H_2, $FLOX/CH_4$, and N_2O_4/A-50 propellants.

For most engine start modes, liquid propellant must be oriented over the propellant feed lines at engine ignition to insure safe and reliable engine starts. This orientation requirement can be accomplished in several ways. In this analysis it was considered passively with surface tension screen devices and actively with the ACS or ullaging rockets. The engine starting mode characteristics and requirements are also an important consideration in sizing and designing a propellant utilization and pressurization system. The pressurization thermodynamics in particular are sensitive to the method of ullage positioning used and the engine start modes.

1.3.1 Summary

A comparison of the propulsion module weight of the idle-mode-start system to that of liquid-containment and ullage rocket systems is presented in Table 5. The system weight differences for all but the hydrogen propellants are small. The pressurization requirements for the hydrogen system using only a containment device are so great, however, that, for this mission profile, depending on a containment system does not appear to be a good approach. A containment system in conjunction with another start mode may be advantageous. The weight of a containment screen was estimated to be 3 lb per tank based on Agena experience.

External ullaging was also evaluated as a start mode. Since the vehicle has adequately sized attitude control system thrusters, they were utilized to provide ullaging thrust. There are two 3 lb-thrust nitrogen gas nozzles in each axial direction providing 6 lb of thrust for ullaging. The time required to move all of the propellant from an assumed upper position in the tanks to the bottom position in the tanks varies from 9 sec for the first burn to 18 sec for the last burn. By the utilization of this ullaging method the penalty is only 6 lb over the idle-mode start thus providing a very effective alternative to the tank-head-idle mode method of operation.

Table 5

ENGINE START MODE EFFECTS

CONCEPT	PROPELLANT	TANK PRESSURE BUILDUP	PROPULSION MODULE WEIGHT (LB)	Δ
IDLE-MODE START	F_2/H_2	WITH WARM PRESSURANT GAS	8,779	(0)
	$FLOX/CH_4$		9,200	(0)
	N_2O_4/A-50		10,471	(0)
CONTAINMENT DEVICE (NO IDLE-MODE START)	F_2/H_2	WITH PRESSURANT GAS AT PROPELLANT SATURATION TEMPERATURE	9,361	(+582)
	$FLOX/CH_4$		9,241	(+41)
	N_2O_4/A-50		10,470	(-1)
ULLAGE ROCKETS (NO IDLE-MODE START)	F_2/H_2	WITH WARM PRESSURANT GAS	8,785	(+6)
	$FLOX/CH_4$		9,206	(+6)
	N_2O_4/A-50		10,477	(+6)

1.3.2 Analysis

The baseline analyses previously conducted assumed an idle mode start which means the engine starts on either liquid or vapor. The pressurization gas is not introduced until after the engine is started and the liquid is settled (Fig. 12a). The two other start mode conditions studied under this sensitivity analysis are as follows:

a. A liquid containment device (screen) is used to assure liquid availability for engine start. There is no positive ullage orientation before starting; i. e., pressurization gas could initially be injected directly into liquid (Fig. 12b).

b. Ullage rockets are used to orient the ullage before gas is introduced, therefore gas is never injected directly into liquid. Also, liquid is assured for engine start (Fig. 12c).

The ullage control system used is important to the pressurization system thermodynamics because with passive control methods the propellant can be covering the pressurant inlet diffuser before engine ignition and ullage positioning. Since the gas would then be injected directly into liquid it would be immediately cooled nearly to the propellant saturation temperature.

The Thermal-Pressurization program was used to analyze the passive liquid-containment system (Fig. 12b) in which the tank is pressurized before the engine is started and the pressurant temperature is equal to the propellant saturation temperature. In this case it is possible to have liquid covering the pressurant injector and therefore heated gas may be injected directly into the propellant. The transient pressure response which would result from immediate propellant vaporization was not computed. However, all energy introduced into the tanks by gas was accounted for in every balance calculation.

In the active ullage control system (Fig. 12c), the analysis was conducted in which the tank was pressurized with an inlet pressurant temperature of 350°R (for all propellants except N_2O_4/A-50) up to the operating pressure, and then was increased to the values shown in Table 6. The initial pressurant inlet temperature for N_2O_4/A-50 was 530°R.

IDLE MODE START

(Baseline)

Pressurizing gas injection started after propellant expulsion is started.

Engine starts on liquid or vapor.

PROPELLANT CONTAINMENT SYSTEM

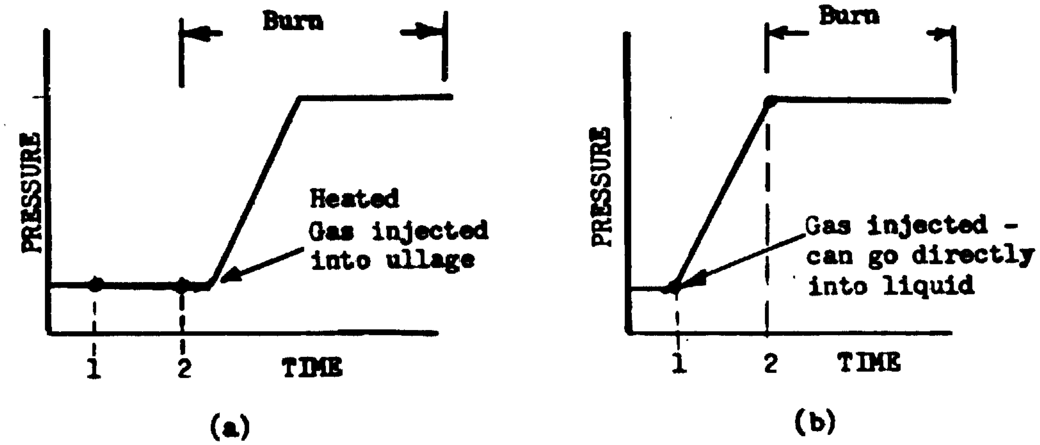
Tank pressure increased to operating level before expelling propellant.

Engine starts on liquid.

LIQUID SETTLING WITH ULLAGE ROCKET

Tank pressure increased to operating level before expelling propellant.

Engine starts on liquid.



Events:
1 - Initiate start sequence
2 - Engine ignition

Fig. 12 Engine Start Mode Sensitivity Analysis

Table 6

ENGINE START MODE
SENSITIVITY ANALYSIS

Case	Prop.	P (psia)	Insulation Thickness (in.)	W_{Vapor} (lb)	$W_{Pressure}$ (lb)	$W_{Insulation}$ (lb)	W_{Tank} (lb)	Σ W (lb)	$T_{Pressure}$* (°R) (1)	$T_{Pressure}$** (°R) (2)
1	F_2	173	1.25	33.4	24.9	26.0	78.1	162	170	650
	H_2	159	4.	39.8	279.7	233.3	256.3	809	50	660
	FLOX	202	1.5	39.0	39.3	33.3	86.7	198	180	435
	CH_4	285	1.625	23.4	19.1	25.3	79.6	147	230	480
	N_2O_4	165	.25	3.4	39.9	7.9	77.6	129	534	650
	A-50	165	.25	.26	30.3	6.8	73.5	111	534	650
2	F_2	155	1.25	33.9	23.8	26.0	77.3	161	350	650
	H_2	155	3.0	66.9	63.7	183.8	266.4	581	350	660
	FLOX	187	1.5	38.7	38.5	33.1	83.4	194	350	435
	CH_4	285	1.625	23.4	19.1	25.3	79.6	147	350	480
	N_2O_4	165	.25	3.4	41.5	7.9	77.6	130	530	650
	A-50	165	.25	.25	32.5	6.8	73.5	113	530	650

*$T_{Pressure\ (1)}$ = Temperature of pressurant before engine start
**$T_{Pressure\ (2)}$ = Temperature of pressurant during engine burn

Cases:
1 — Passive liquid containment method [Fig.12 (b)]
2 — Active ullage positioning method — applicable to both idle mode start [Fig.12 (a)] and ullage rocket methods [Fig.12 (c)]

Thermodynamic system weights for both systems are also presented in Table 5. It can be noted that with the exception of the hydrogen system, differences in weights for the two methods are slight. With the passive liquid-containment method for hydrogen, the possible injection of pressurant directly into the liquid gives a significant weight penalty.

1.4 THRUST AND CHAMBER PRESSURE SENSITIVITY

The Mars Orbiter vehicle as defined for study in Phase I utilizes an 8,000 lb thrust engine with a chamber pressure of 100 psi for the pressure-fed cases. This is not necessarily optimum, and consequently variations of these combinations were studied. Thrust levels were varied from 1000 to 8000 lb and chamber pressure was varied from 50 to 150 psia. The propellants investigated were F_2/H_2, $FLOX/CH_4$, and N_2O_4/A-50. These variations in thrust and chamber pressure have a significant effect on other parameters such as engine weights and specific impulse values. The sensitivity to burn duration was assessed in the thermodynamic analysis. The sensitivity to gravity losses was incorporated in the performance analysis as a function of thrust level and propellant combination. The effect on the overall vehicle system was determined by conducting an overall system optimization in which insulation thickness, tank pressure, and system weights are defined.

1.4.1 Summary

The effects of thrust and chamber pressure on total weight of a pressure-fed Mars Orbiter propulsion stage are summarized in Fig. 13. For all three propellants analyzed the lightest weight systems are achieved at thrusts between 4000 to 6000 lb. Increasing thrust to 8000 lb introduces a slight weight penalty, while reducing thrust to 1000 lb introduces a large weight penalty. In the N_2O_4/A-50 and $FLOX/CH_4$ cases the performance increases with chamber pressure for all cases studied. For the F_2/H_2 cases the 100 psia chamber pressure systems yield the best performance. The increase in specific impulse and decreased engine weight for the 150 psia cases are not enough to offset the vapor, pressurization, and tank weight penalties for this case.

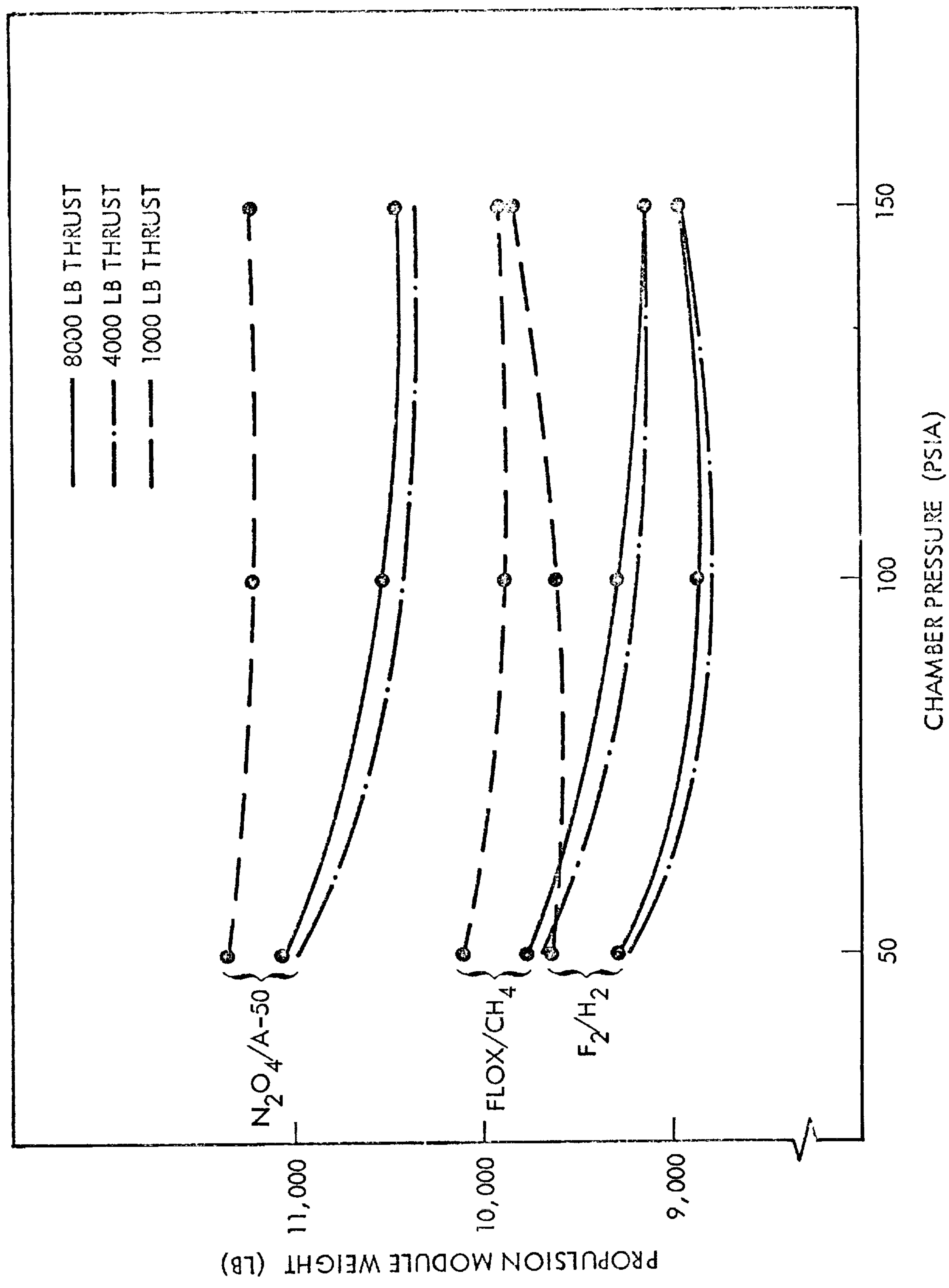


Fig. 13 Thrust and P_C Sensitivity Summary

1.4.2 Propulsion Data

The basic propulsion data for the analysis are tabulated in Table 7. Included are combinations of thrust, chamber pressure, engine weights, and specific impulse for the various propellant combinations, based on pressure-fed systems with an expansion ratio of 100/1. These data are also plotted in Figs. 14, 15, 16, and 17.

The engines were all assumed to be ablatively/radiatively cooled pressure-fed systems and the following pressure drops were used for injector and line losses:

F_2/H_2	55 psia
$FLOX/CH_4$	60 psia
N_2O_4/A-50	65 psia

Another item of consideration is the burn time since it affects both the pressurant expulsion cycle and engine life. The following nominal burn times were used in the analysis:

Thrust Level, lb		Nominal Burn Time, sec			
Primary	Secondary	Midcourse 1	Midcourse 2	Orbit Inject	Orbit Trim
8000	1000	90	90	330	55
4000	1000	90	90	660	55
1000	1000	90	90	2640	55

1.4.2 Thermodynamics

The computer code used for the pressure-fed analysis was updated to include new National Bureau of Standards (NBS) hydrogen properties and a modified computation procedure. In addition, to conserve computer time, a single pressurant inlet temperature was assumed based on the optimum temperatures obtained from the analyses reported in the Phase I final report. The system pressure drop values assumed for

Table 7

PROPULSION SYSTEM CHARACTERISTICS

Case	Chamber Pressure, psia	Thrust, lb		Engine Weight, lb	Isp, LBF/LBM/sec					
					F_2/H_2		$FLOX/CH_4$		N_2O_4/A-50	
		Primary	Secondary		Primary	Secondary	Primary	Secondary	Primary	Secondary
1	150	1000	1000	35	437	437	385	385	326	326
2	150	8000	1000	243	447	437	392	385	331	326
3	100	1000	1000	47	432	432	381	381	323	323
4	100	4000	1000	152	438	432	385	381	326	323
5	50	1000	1000	68	423	423	373	373	319	319
6	50	8000	1000	523	430	423	379	373	323	319
7	100	8000	1000	313	442	432	387	381	328	323

Expansion Ratio is 100/1

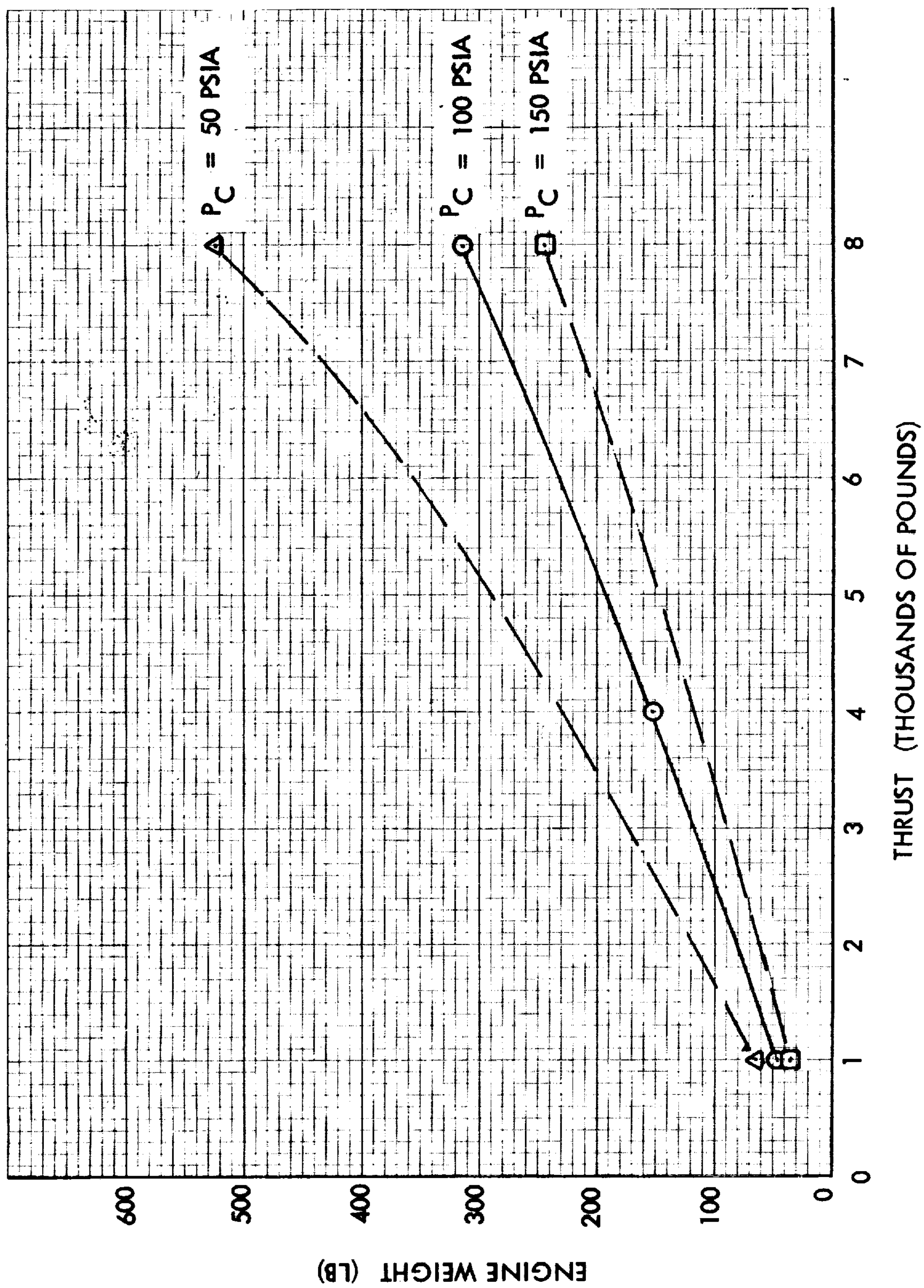


Fig. 14 Engine Weight Variation With Thrust and Chamber Pressure

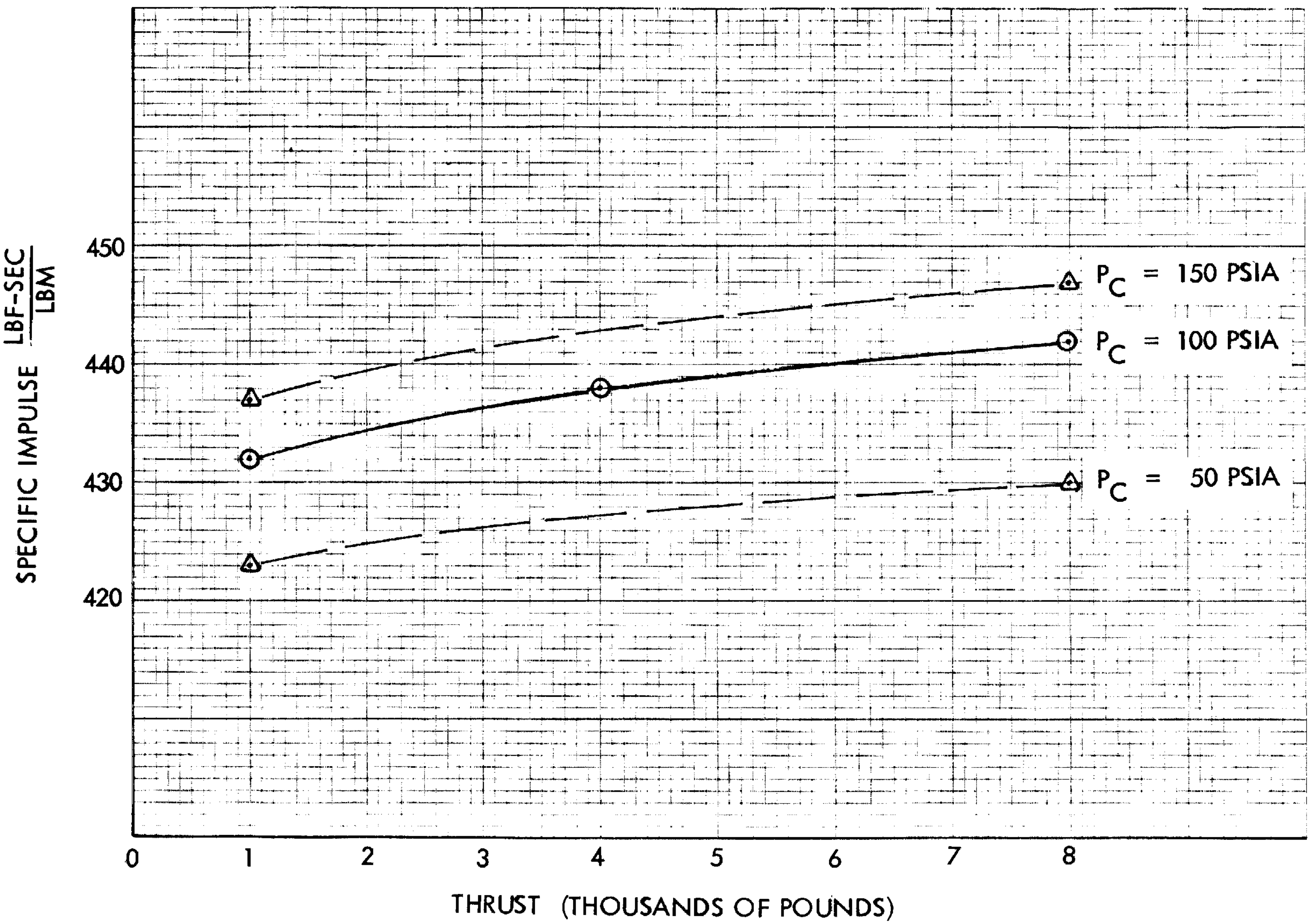


Fig. 15 F_2/H_2 Specific Impulse Variation With Thrust and Chamber Pressure

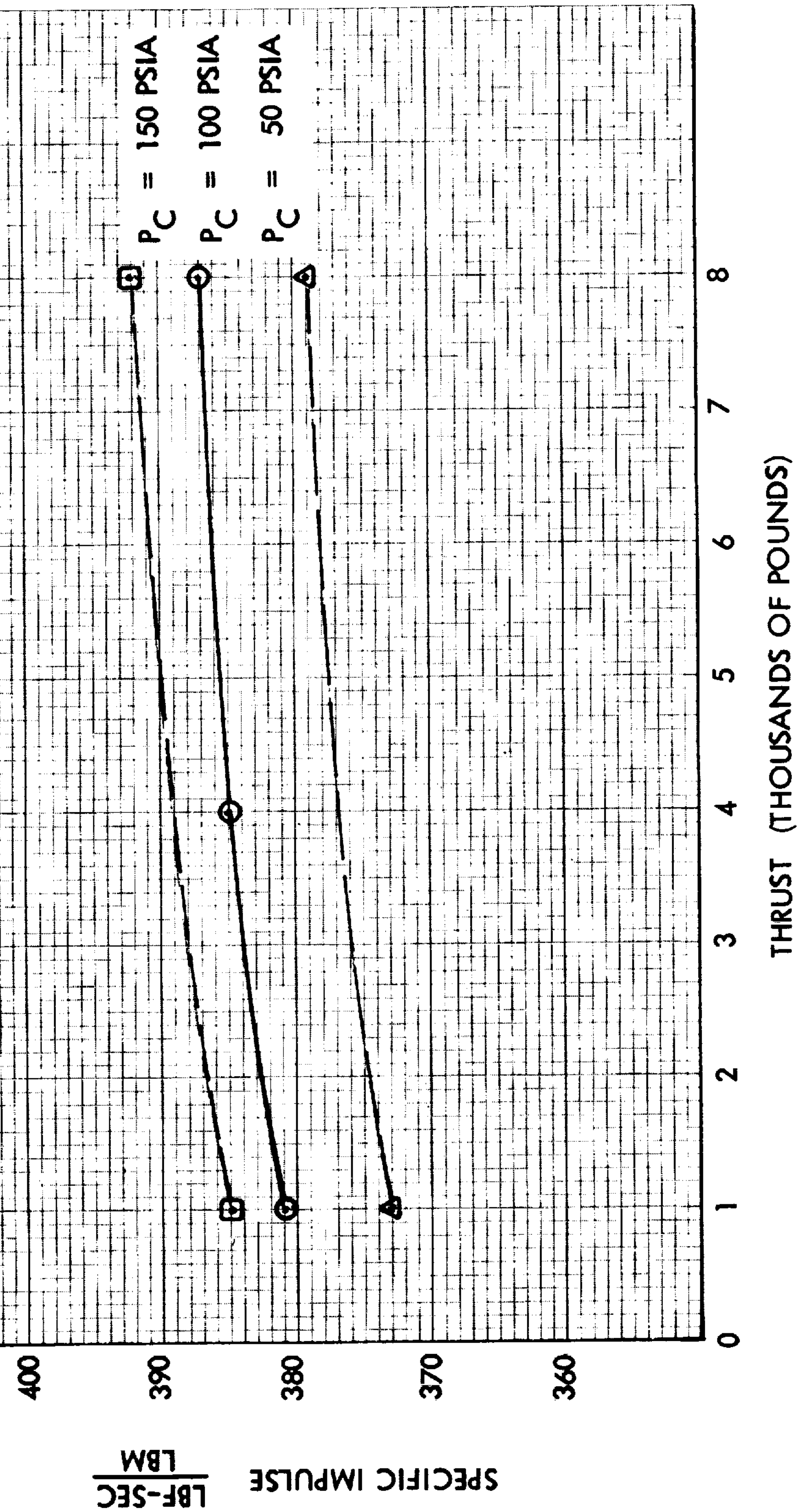


Fig. 16 FLOX/CH_4 Specific Impulse Variation With Thrust and Chamber Pressure

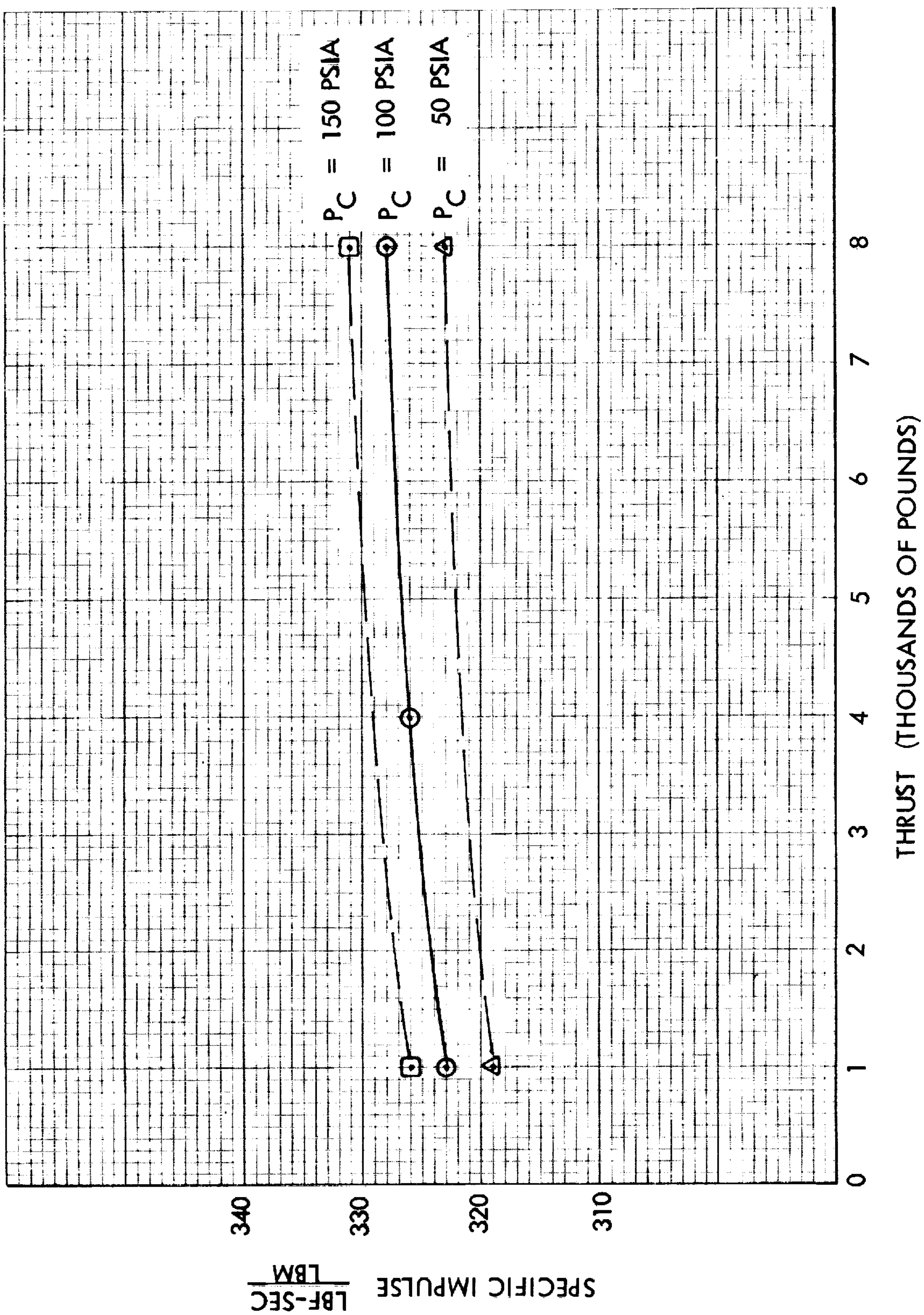


Fig. 17 N_2O_4/A-50 Specific Impulse Variation With Thrust and Chamber Pressure each propellant were reduced from the values used in the original baseline study because of the new engine assumption. This results in changes in the minimum total tank pressures. This study assumed a two-tank F_2 configuration instead of a 4-tank configuration as originally assumed for the pressure-fed systems. However, the baseline conditions were re-run for a 2-tank configuration which gives a common basis for comparison.

Changing the expulsion duration as required by changes in thrust level affects the pressurant requirements slightly due to changes in the total heat transfer from the pressurizing gas to the tank wall. However, hydrogen was the only propellant where these ullage heating effects were significant enough to be detectable, as shown in Table 8. The weight changes of the other propellants analyzed resulted from total tank pressure adjustments caused by the different chamber pressures.

1.4.3 Systems and Performance

Based on the various engine systems and propellant-sensitive parameters new vehicles were defined and scaling laws developed for their evaluation in a performance analysis. This performance analysis was further enhanced by incorporating the gravity losses associated with the 6294 ft/sec orbit injection maneuver in order to define the thrust sensitivity as accurately as possible. The actual gravity losses considered are shown in the following table:

Chamber Pressure psia	Thrust lb	Gravity Losses, ft/sec		
		N_2O_4/A-50	FLOX/CH_4	F_2H_2
150	1000	637	719	816
150	8000	37	37	41
100	1000	631	711	799
100	4000	130	139	153
50	1000	625	701	784
50	8000	39	40	42
100	8000	37	37	40

Table 8

THRUST AND CHAMBER PRESSURE SENSITIVITY ANALYSIS

Case	Prop.	P (psia)	Insulation Thickness (in.)	W_{Vapor} (lb)	$W_{Pressure}$ (lb)	$W_{Insulation}$ (lb)	W_{Tank} (lb)	$\sum W$ (lb)	P_c	Thrust
1	F_2	208	1.25	37.8	32.3	26	84	180	150	1,000
	H_2	205	3.0	78.1	115	184.7	340.5	718		
	FLOX	215	2.0	32.7	46.2	44.2	84.0	207		
	CH_4	323	1.625	23.4	24.2	25.3	82.5	155		
	N_2O_4	215	0.25	3.5	54	7.8	85	150		
	A-50	215	0.25	0.26	42	6.8	78	127		
2	F_2	208	1.25	37.7	32.3	26	84	159	150	8,000
	H_2	205	3.0	78.7	100	184.7	340.5	703		
	FLOX	215	2.0	32.7	46.2	44.2	84.0	207		
	CH_4	323	1.625	23.4	24.2	25.3	82.5	155		
	N_2O_4	215	0.25	3.5	54.1	7.9	85.7	151		
	A-50	215	0.25	0.26	42.1	6.7	78.2	127		
3	F_2	168	1.25	33.5	24.1	26.0	77.3	161	100	1,000
	H_2	155	3.0	66.6	82	183.8	266.4	599		
	FLOX	187	1.5	38.7	38.5	33.1	83.4	194		
	CH_4	285	1.625	23.4	19.1	25.3	79.6	147		
	N_2O_4	165	0.25	3.4	41.5	7.9	77.6	130		
	A-50	165	0.25	0.26	32.4	6.8	73.5	113		
4	F_2	167.7	1.25	33.5	24.1	26.0	77.3	161	100	4,000
	H_2	155	3.0	65.5	73	183.8	266.4	588		
	FLOX	187	1.5	38.7	38.5	33.1	83.4	194		
	CH_4	285	1.625	23.4	19.1	25.3	79.6	147		
	N_2O_4	165	0.25	3.4	41.5	7.9	77.6	130		
	A-50	165	0.25	0.25	32.4	6.8	73.5	113		
5	F_2	125	1.25	29.1	16.2	26.0	77.3	149	50	1,000
	H_2	112	4.0	45.0	52.4	226.0	187	510		
	FLOX	120	2.0	32.1	25.9	43.3	77.8	179		
	CH_4	233	1.625	21.9	13.4	24.9	73.2	133		
	N_2O_4	115	0.25	3.3	28.8	7.9	77.6	118		
	A-50	115	0.25	0.25	22.7	6.8	73.5	103		
6	F_2	125	1.25	29.1	16.2	26.0	77.3	149	50	8,000
	H_2	111	4.0	43.8	43.0	225.4	184.6	498		
	FLOX	120	2.0	32.1	25.9	43.3	77.8	179		
	CH_4	233	1.625	21.9	13.4	24.9	73.2	133		
	N_2O_4	115	0.25	3.3	28.8	7.9	77.6	118		
	A-50	115	0.25	0.25	22.7	6.8	73.5	103		
7	F_2	155	1.25	33.9	23.8	26.0	77.3	161	100	8,000
	H_2	155	3.0	66.9	63.7	183.8	266.4	581		
	FLOX	187	1.5	38.7	38.5	33.1	83.4	194		
	CH_4	285	1.625	23.4	19.1	25.3	79.6	147		
	N_2O_4	165	0.25	3.4	41.5	7.9	77.6	130		
	A-50	165	0.25	0.25	32.5	6.8	73.5	113		

The propulsion module weights for the N_2O_4/A-50, FLOX/CH_4 and F_2/H_2 stages are shown in Figs. 18 through 20, respectively. The computed data points are shown and the curves are "best guess" fairings.

1.5 PROPELLANT LEAKAGE EFFECTS

Propellant leakage has an effect on both propellant loss and heat transfer to the propellant tank. There is a considerable variation and effect of these characteristics for the different propellants. An assessment was made of the actual propellant that could be lost due to leakage for the Mars Orbiter vehicle on its 205-day mission and the effect that propellants in the feed lines have on heat transfer to the propellant tanks. Consideration was given to reducing both the actual losses and thermal effects by design and operational consideration. For the F_2/H_2 and FLOX/CH_4 systems the vehicle is in sun-on-capsule orientation and for the N_2O_4/A-50 propellants in sun-on-tank orientation.

The propulsion systems with N_2O_4/A-50 and FLOX/CH_4 have engine shutoff valves only and no tank shutoff valves. For F_2/H_2 both tank shutoff and engine shutoff valves are installed for the H_2. Consequently valve leakage will have a thermodynamic effect on the F_2/H_2 system only.

1.5.1 Summary

Propellant leakage through feed line valves is dependent upon many factors. It is an area requiring intensive development for the propellants analyzed since only very limited data exists. Tested leakage rates are relatively small with values of 5 lb total or less for hydrogen on a Mars Orbiter vehicle. This value can be reduced still further with valves fabricated to closer tolerances at increased cost.

The effect of having propellant trapped in the feed line is of concern only for the hydrogen design. Should this condition exist, the design prevents over-pressurization by

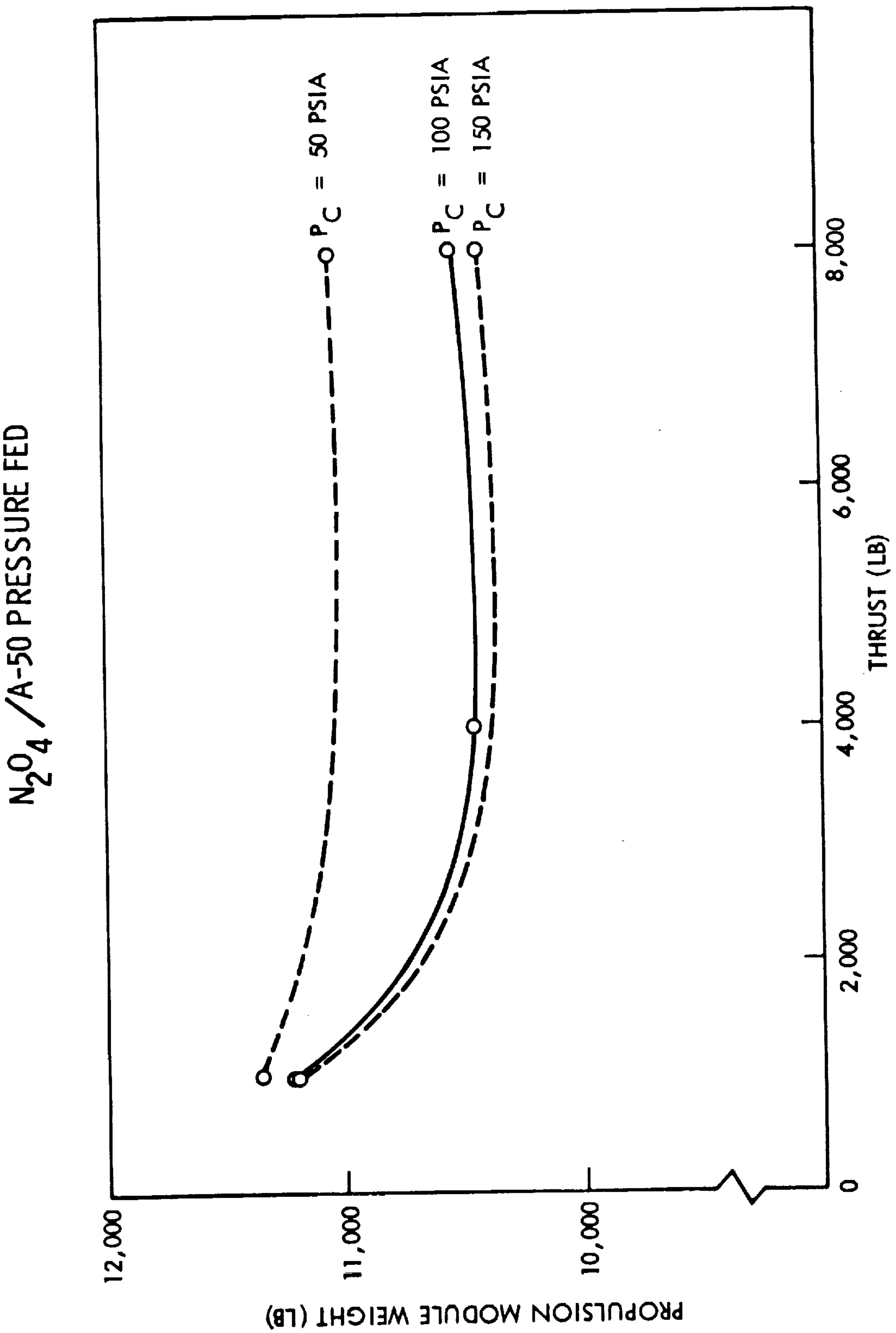


Fig. 18 Thurst and P_c Sensitivity Results – N_2O_4/A-50

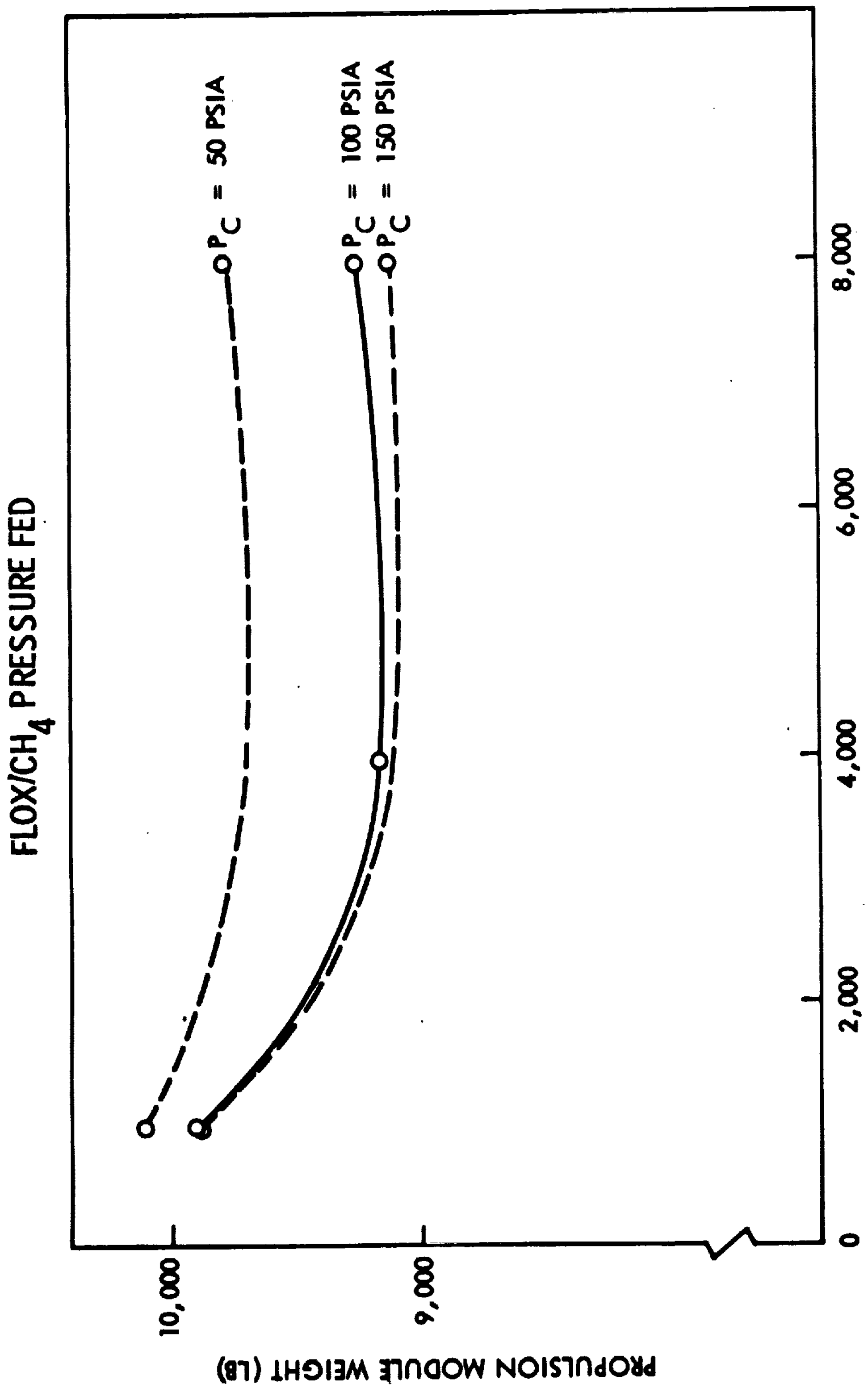


Fig. 19 Thrust and P_c Sensitivity Results – FLOX/CH_4

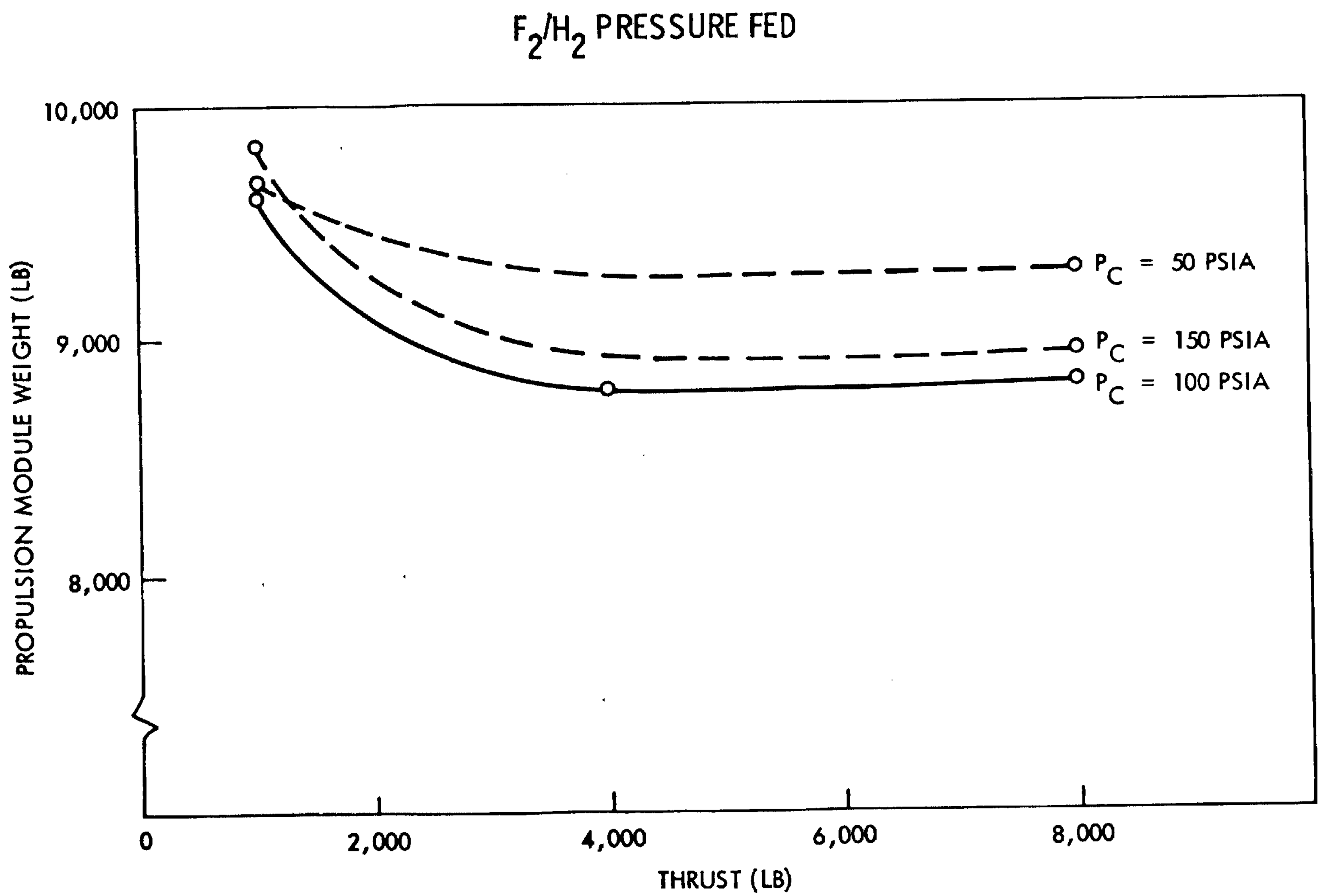


Fig. 20 Thrust and P_c Sensitivity Results – F_2/H_2

allowing propellant to return to the tank by means of a bleed line. After equilibrium is reached, vapor in the line increases the conductivity of the propellant feed line by approximately 34 percent. Since all the plumbing penetration represents less than 15 percent of the total heating this effect increases the total heat leak by less than 5 percent.

1.5.2 Mechanical Systems

The leakage of valves is a complex subject with many variables. Very little directly applicable data exists for determining valve leakage. Discussions were held with the engine companies and with vendors. Considerable data was evaluated in this process. In addition, an analysis was made by correlating existing valve data to the Mars Orbiter operating conditions. A J. C. Carter flight-type F_2 valve was used for this correlation. A leakage rate of 2 to 20 Standard Cubic Centimeters of helium per hour was obtained through a 1-in. diameter valve at 520°R and with an upstream pressure of 50 psia and vacuum downstream during a series of development tests. At the conclusion of these development tests an acceptance test was conducted. The valve seat now had some scratches and some contamination was present. The leakage rate had consequently increased to 170 Standard Cubic Centimeters of Helium per hour under the same conditions. This value which can be construed as an upper limit was used as the basis for correlating valve leakage to the Mars Orbiter vehicle. Assuming this value as the basis for the correlation, the following leakage was determined for the total mission.

Propellant	Valve Size, In.	Leakage, lb
N_2O_4	1	0.5
A-50	1-1/2	0.6
FLOX	1-1/2	2.9
CH_4	1-1/2	2.8
F_2	1-1/2	2.2
H_2	1-1/2	4.6

These results are based on actual tank pressures in the Mars Orbiter vehicle and consider only gaseous leakage which would normally occur. If liquid leakage exists, these values would be smaller.

The correlation equations for the analysis were obtained from NBS TN 355 (8/1/67) "Correlation for Predicting Leakage Through Closed Valves."

The equations used assume:

1. Isothermal flow
2. Velocity of approach is negligible
3. No change of leak path geometry with pressure or temperature
4. Subsonic flow
5. Specific heat of gases do not differ appreciably .
6. Friction factor does not vary widely

The type of flow assumed can be:

1. Free molecular
2. Subsonic continuum
3. Turbulent incompressible
4. Transition flow through orifices
5. Isothermal compressible
6. Critical flow through rounded orifices, nozzles and long channels.

In addition, the specific test data used were corrected for the appropriate propellant characteristics, valve size, and upstream pressure with the following equation:

$$\frac{\dot{w}_A}{\dot{w}_B} = \frac{P_A}{P_B}\left(\frac{R_B\, T_B}{R_A\, T_A}\right)^{1/2}\left(\frac{D_A}{D_B}\right)^2$$

where

$\dot{w}$ = mass flow

P = gas pressure

R = gas constant

T = gas temperature

D = valve diameter

Leakage which is usually measured with either nitrogen or helium could be decreased by an order of magnitude with some development and with special processing. Douglas Aircraft Company in Report NASA CR 72063, DAC-60599, dated July 1967, states that fluorine leakage rates as low as 5×10^{-10} lb per second or .009 lb for a 205 day mission can be obtained using surfaces lapped to one micro-inch roughness with seat stresses of about 10,000 psi. Rocketdyne estimates that leakage rates of 0.1 to 3.0 lb per year can be obtained for fluorine in these applications.

Leakage of valves, although difficult to determine exactly is rather small and should not cause significant mission problems.

1.5.3 Thermodynamics

The effect of valve leakage on heat transfer into the propellant tanks was investigated. The thermal analysis involved determining the increase in conduction heat transfer along the feed lines as a function of propellant vapor conductivity and pressure. Leakage affects heat transfer only to the H_2 tank which utilizes both a tank shut-off and an engine valve. The feed line then becomes a cavity in which propellant vapor which leaks can be trapped.

All oxidizer systems employ only one valve in the feed line. Leakage of vapor through the valve is assumed to vent through the engine to space; thus, there is no trapped vapor downstream of the valve and heat transfer due to leakage is considered negligible.

The H_2 shutdown sequence will presumably allow post flow cooling in which case the tank valve is shut off first; then, after emptying the feed line through the engine the engine valve is closed. The fuel line at this time may be nearly evacuated or partially filled with fuel vapor.

After the transient condition of line boilout occurs, the fuel line can be full or soon be filled with vapor if the tank valve leaks. The feed line conduction heat transfer would increase relative to an evacuated line due to conduction through the vapor as well as the line wall. An analysis was made assuming a varying temperature along the feed line and/or vapor while holding the outer insulation surface and the hot boundary (engine) temperatures constant. A 4-ft long, 2-1/2 in. diameter, stainless steel feed line with a wall thickness of 0.010 in. was assumed, insulated with the equivalent of 1 in. of multilayer insulation. The following table presents the percent increase in feed line conduction heat leak anticipated for two gas conductivities using this geometry:

Conductivity Btu/ft-hr°R	Percent Increase in Heat Leak
0.01	23
0.10	34

These conductivity values span the region of conductivities for all propellants analyzed and for pressures up to about 10 atmospheres as shown in Fig. 21. In addition, the line size assumed is larger than any of the designs in this study and the wall thickness is a minimum, therefore, the conduction heat leak increase due to the presence of vapor would actually be slightly less than predicted. Thus, the analysis made represents the greatest possible change in conduction heat leak for a vapor filled line.

The feed line heat leaks are most significant for sun-shielded or sun-on-capsule orientations in that the heat leak is a greater percentage of the total heat input than with a sun-on-tank orientation. However, plumbing penetrations contribute less than 10 to 15 percent of the total heating for these configurations so a 23 to 34 percent increase in feed line heat leak would cause only a 2 to 5 percent increase in the total heat leak. This increase in heating would have only a slight effect on system weight.

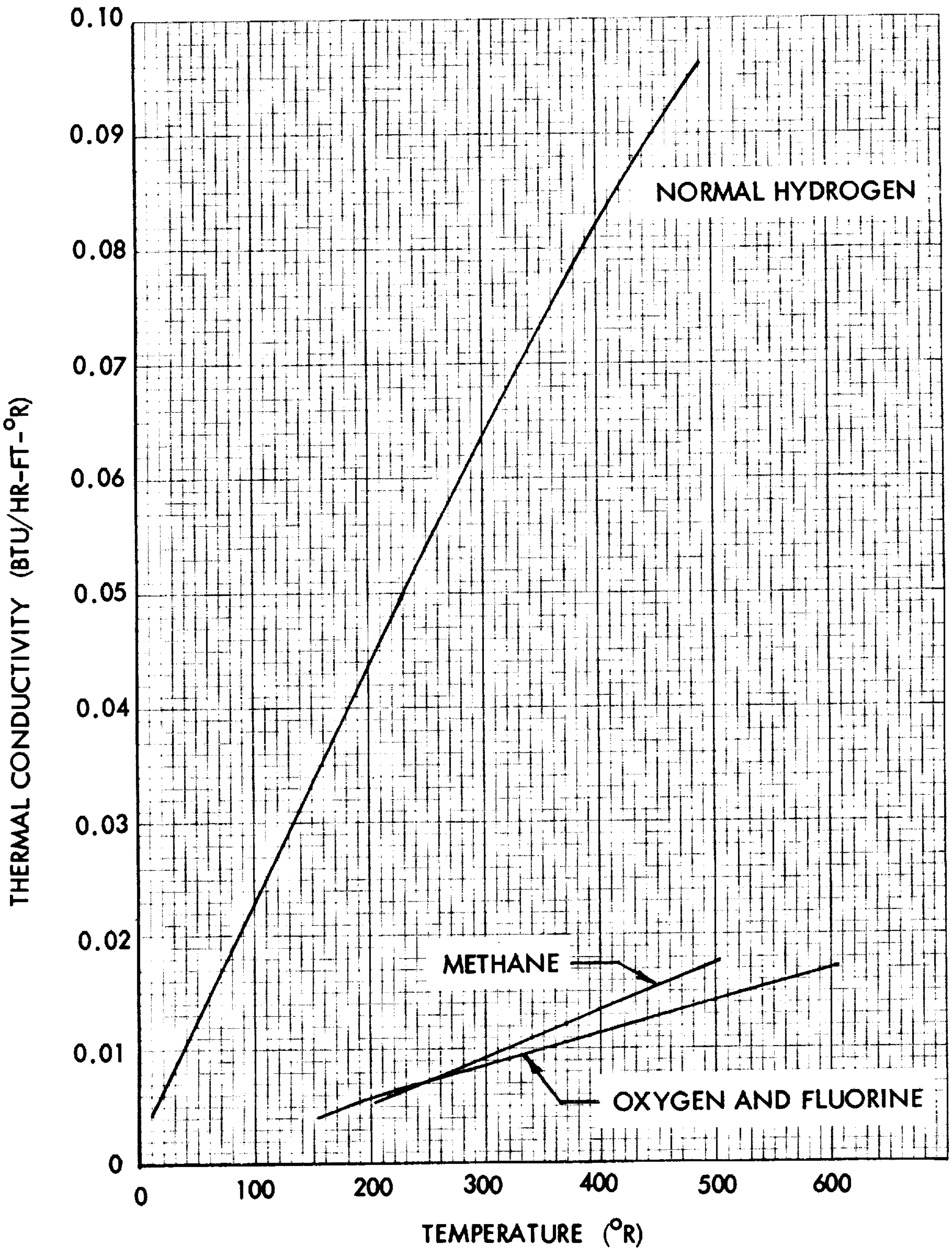


Fig. 21 Thermal Conductivity of Propellant Vapors

Any propellant leakage could increase a "locked up" feed line pressure to equal or greater than tank pressure. Such a pressure rise would stop tank valve leakage or at least permit equilibrium conditions to be reached by bleeding vapor back into the tank as required. Valve leakage at the pump or engine interface would slightly reduce the heat leak.

The overall effect of increased heat leak and vapor pressure buildup in the propellant feed lines appears to be negligible. A check valve bleeding back into the tank should be incorporated for redundancy in maintaining a reasonable pressure level in the feed line.

An area of uncertainty in the effect of leakage is the characteristics of vaporization and freezing of propellant when expanding through leakage paths in the valve. Freezing and subsequent sublimation is expected in these areas, but relatively little is known of the possible effects on valve operation. Research in this area is recommended.

A detailed study of post burn heat leaks is required with emphasis on post-flow requirements and engine soak back, especially through the feed line.

1.6 DESIGN SIMPLIFICATION

The propulsion system for each propellant combination has a different configuration and different operational cycle. The complexity of these systems is also different. An analysis was made in order to compare the complexity of the various propulsion systems. This task was performed in three phases. In the first phase a search was made for the most simple system, using sun-shaded tanks for the F_2/H_2 and FLOX/CH_4 systems and sun-exposed tanks for the N_2O_4/A-50 system. The second phase assessed the number of functional elements and their operation. The third phase described the operational sequences for each system. The results of the second and third phase efforts are incorporated in Section 2 of this volume.

1.6.1 Summary

The design simplification analysis indicated that the space-storable propellant system need be no more complex than the earth-storable system if the tanks are properly sun-shielded. Prevalves at the propellant tanks are eliminated, but the reduction in component weight approximates the increase in thermal penalty and consequently no net effect on performance is seen.

For the F_2/H_2 system the prevalves can be removed for the F_2 tanks but are still required for the H_2.

<u>System Simplification</u>

Each propulsion system has requirements that demand a different configuration. The simplest system is represented by the Earth storable system shown in Fig. 22. This system is representative of a propulsion system in thermal equilibrium with its environment. One of the things the figure indicates is that the oxidizer tanks are subject to pressurant controlled by regulator RG2 and the fuel is subject to pressurant controlled by regulator RG3. The reason for independent pressure regulation in this study is that the oxidizer and fuel systems are optimized independently and can operate at different pressure levels. One simplfication that could be made, however, is to control the higher pressure with a regulator and the lower pressure with a calibrated orifice. Pressurization valve U6 is used only for ground purge operations of the engine and has no flight functions.

The space storable propulsion system that had been proposed earlier is shown schematically in Fig. 23. Careful analysis indicated that this system could be simplified. For a sun-shielded vehicle, the super-insulation requirements are at the minimum values established, 1/2-in. or less, and the tank operating pressures are also extremely low. Consequently some simplifications were possible without incurring any significant thermal penalties. The revised propulsion schematic is shown in Fig. 24. This configuration is essentially of the same complexity as the Earth storable configuration.

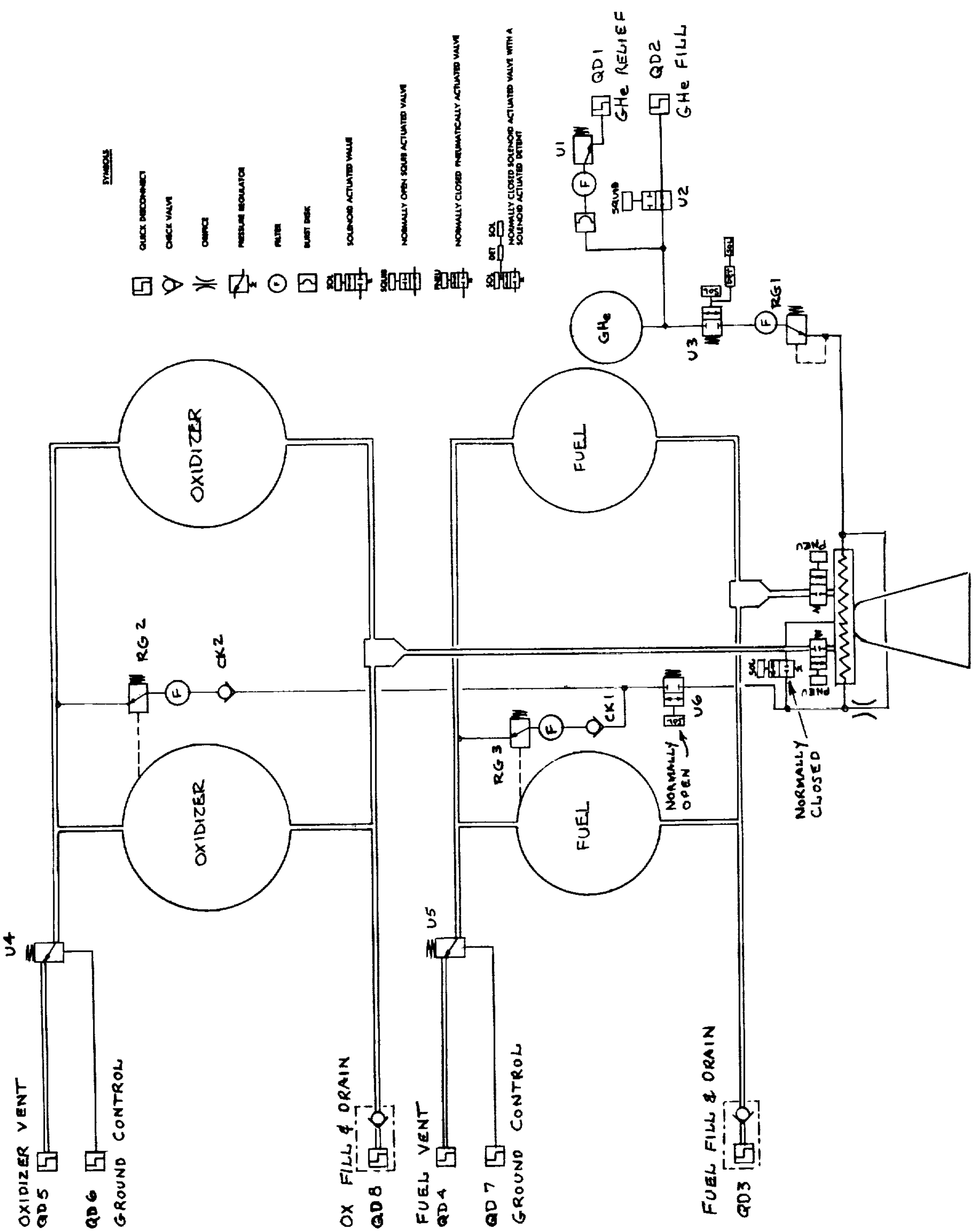


Fig. 22 Mars Orbiter Fluid Systems Schematic – Earth Storable Propellants

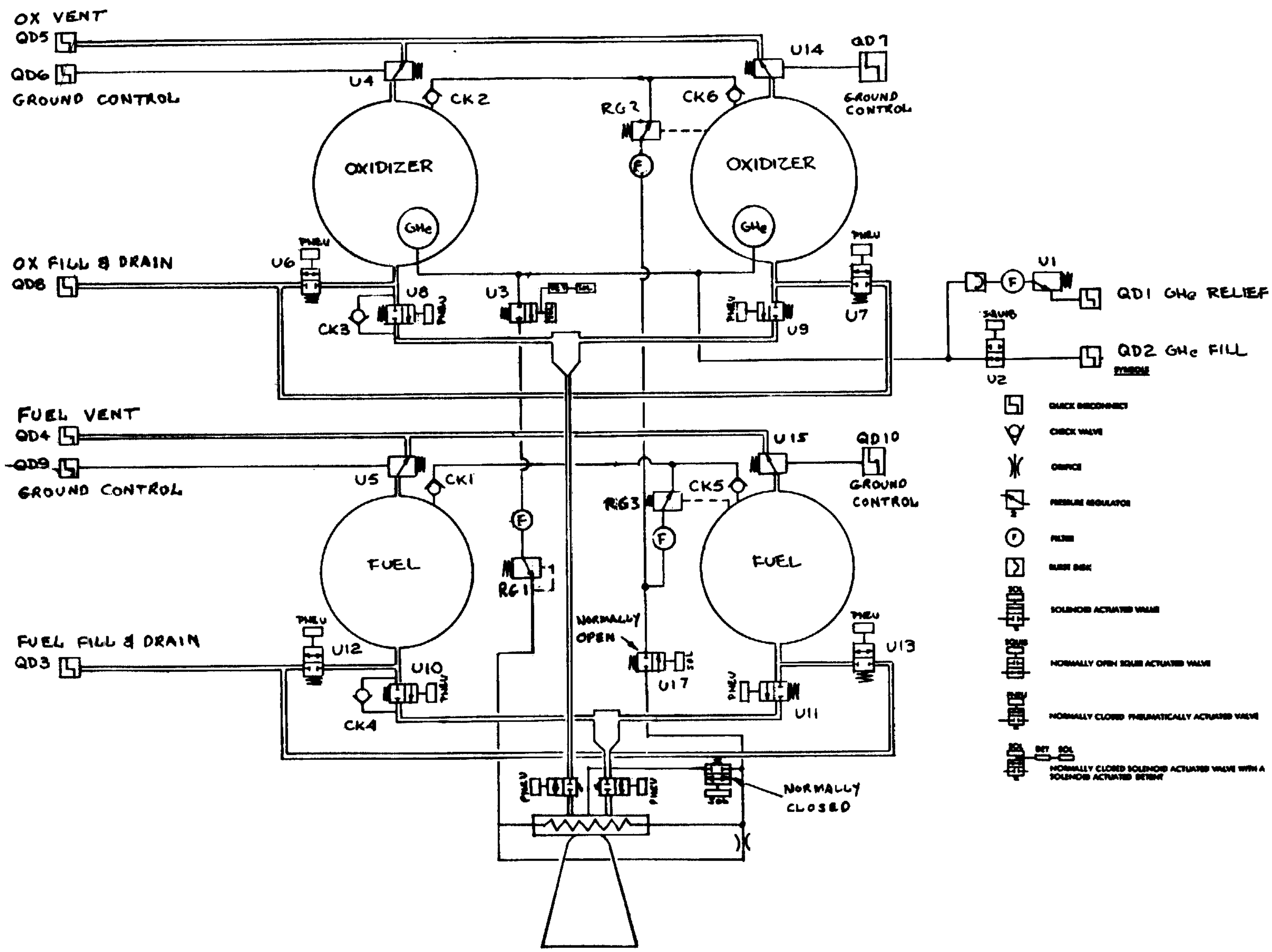


Fig. 23 Mars Orbiter Fluid Systems Schematic — Space Storable Propellants

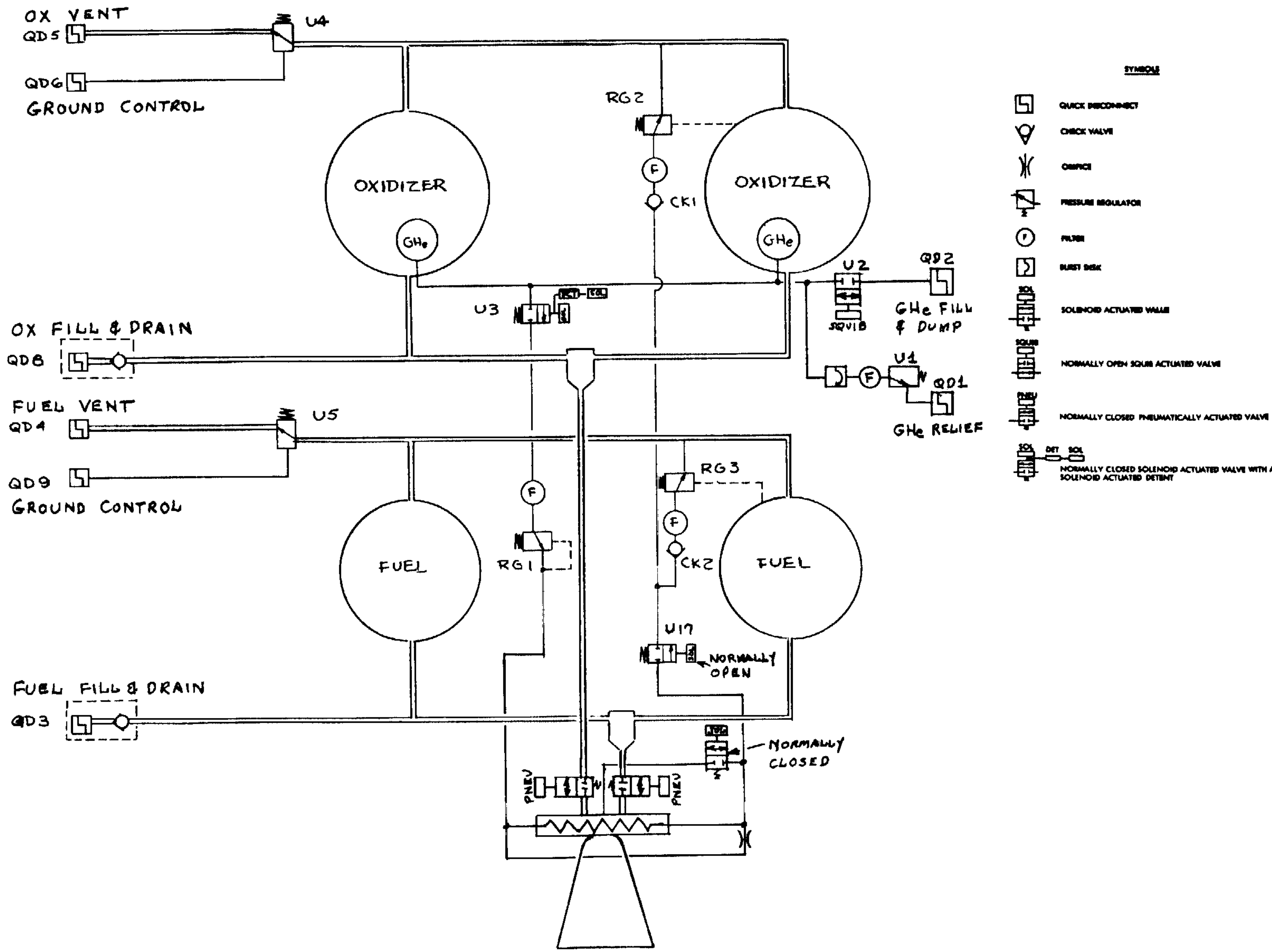


Fig. 24 Mars Orbiter Fluid Systems Schematic (Simplified) – Space Storable Propellants (No Propellant Isolation)

The propulsion configuration for the cryogenic system that had been proposed earlier is shown in Fig. 25. It can be simplified for the F_2 propellants as shown in Fig. 26 to reduce the valve weights by 18 lb. If extensive shadow shielding, sun-orientation control during maneuvers, and the latest in technology are employed, further simplifications could be implemented. For both the space storable and cryogenic systems the pressurizations spheres should be at the environment of the coldest propellant. In this analysis, the pressurization spheres were assumed located in the propellant tanks, but could also be external and enclosed within the propellant tank insulation system.

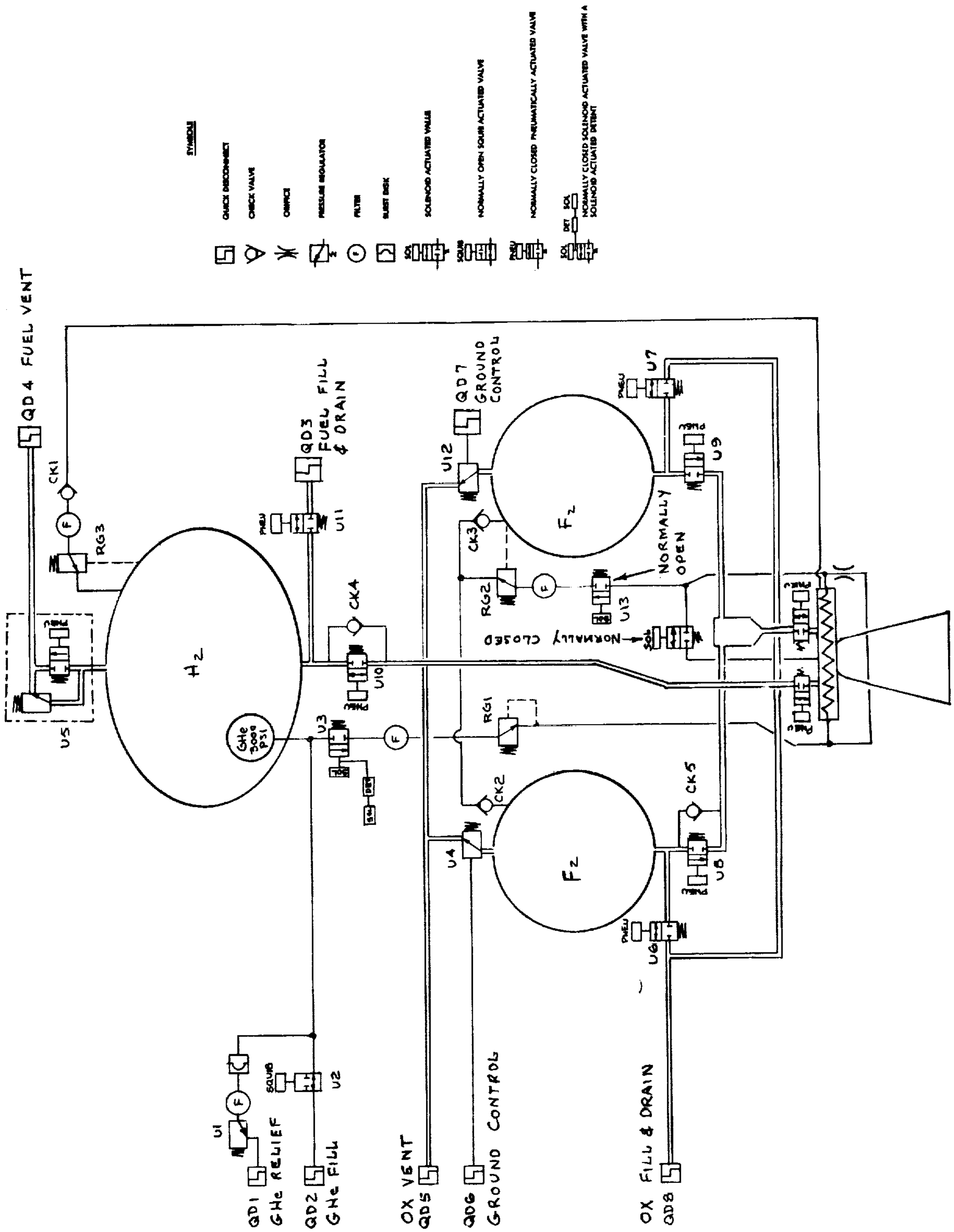


Fig. 25 Mars Orbiter Fluid Systems Schematic – Cryogenic Propellants

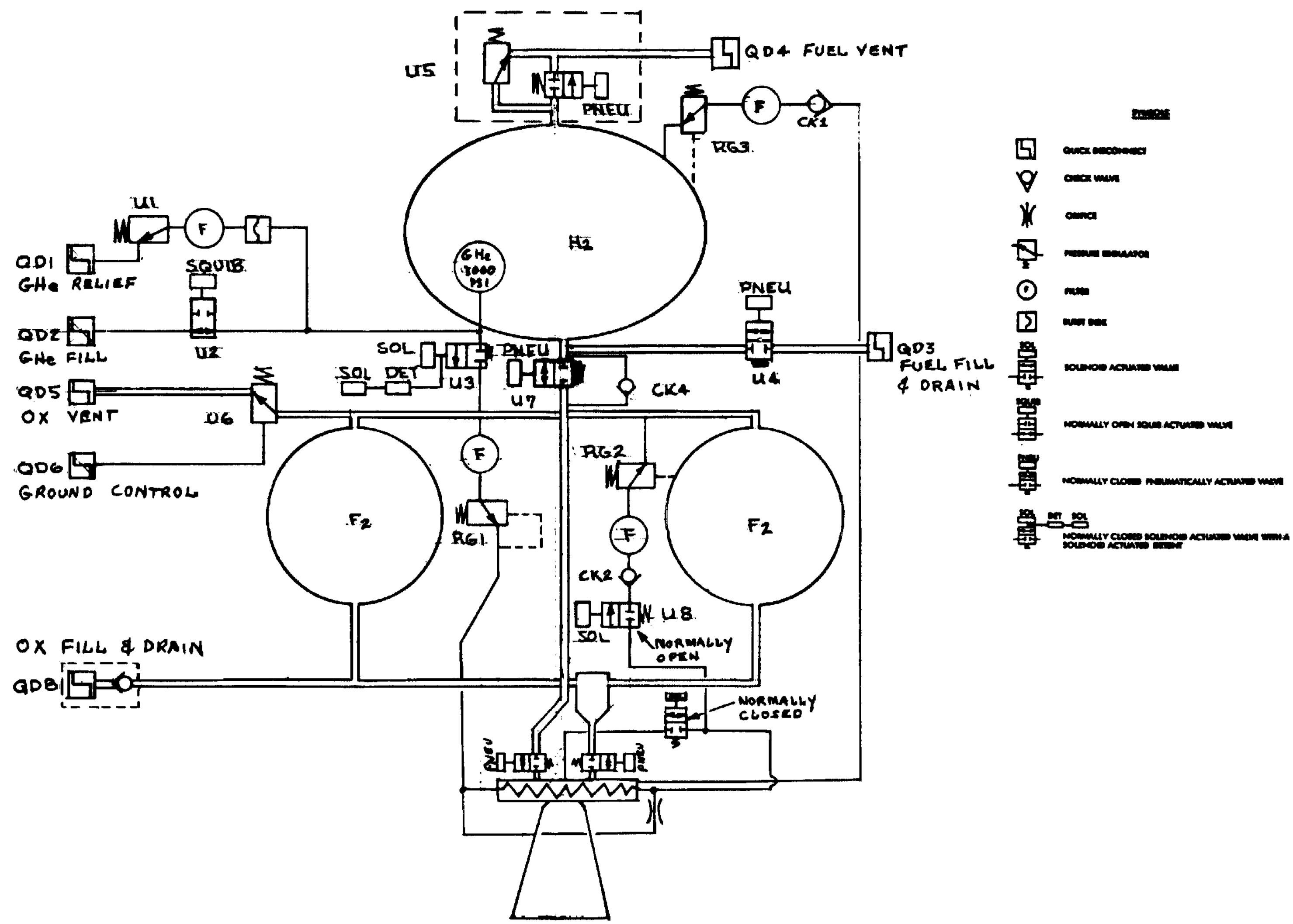


Fig. 26 Mars Orbiter Fluid Systems Schematic – Cryogenic Propellants (NO F_2 Propellant Isolation)

Section 2

ANALYSIS OF SECONDARY VERSUS PRIMARY PROPULSION FOR MINOR ΔV REQUIREMENTS

This section presents the results of a comparison of the performance and complexity of alternate propulsion modes for accomplishing minor ΔV corrections. In particular, the comparison of a primary propulsion system used in the throttled mode versus a separate propulsion system sized for the secondary ΔV requirements was of greatest interest.

The basis for the analysis was as follows:

Mission — Mars Orbiter used in Phase I of NASw-1644

Main Engine — 8,000-lb thrust, pump-fed

Propellants — (1) F_2/H_2, FLOX/CH_4, and N_2O_4/A-50 with common propellants for primary and secondary systems, or
(2) N_2O_4/A-50 for secondary propulsion regardless of propellant in the primary

Maneuvers — (1) 164 ft/sec at 3 days from launch
(2) 164 ft/sec at 165 days from launch
(3) 6,294 ft/sec at 195 days from launch
(4) 328 ft/sec at 205 days from launch

Payload — 8,143 lb to Mars orbit
3,143 lb for final burn (having separated a 5,000-lb lander capsule)

The propulsion modes considered are shown in Table 9.

Note: See page 122 for a summary of this task.

Table 9

PROPULSION MODES

Mode \ Burn	1	2	3	4
A	Primary Throttled	Primary Throttled	Primary	Primary Throttled
B	Common Pro-pellant, Separate Tanks	Common Pro-pellant, Separate Tanks	Primary	Common Pro-pellants, Separate Tanks
C	Common Pro-pellant, Common Tanks	Common Pro-pellant, Common Tanks	Primary	Common Pro-pellant, Common Tanks
D	N_2O_4/A-50 Secondary	N_2O_4/A-50 Secondary	Primary	N_2O_4/A-50 Secondary
E	Primary-Idle	Primary-Idle	Primary	Primary-Idle
F	ACS	ACS	Primary	ACS

The following guidelines were adopted from the cited references for defining a secondary propulsion system:

a. From Ref. 1 – Although odd numbers of chambers could be used, it is more practical to use pairs oriented to given symmetry about one or more of the major control axes.

b. From Ref. 1 – Guidance analyses have shown that either 200- or 400-lb thrust offers reasonable compromise between short burn time and long burn time inaccuracies.

Ref. 1 Voyager Spacecraft System, Phase 1A Task B: Preliminary Design. Propulsion Analysis, General Electric Co., 31 Jan 1966, NASA CR 71500

c. From Ref. 2 — The requirement at low thrust during midcourse maneuver firings is one meter per second minimum ΔV, with a shutdown error of 0.04 meter per second (3σ).

2.1 ANALYSIS APPROACH

The analysis approach used was to:

1. Set up specific propulsion systems requirements and define engine systems in terms of size, weight, and performance.
2. Analyze pressurization system and thermodynamic effects of a single-burn main propulsion system to determine the reduction in primary system weight and complexity over the baseline multiburn, throttled system.
3. Describe the operational sequences and functional parts of competing systems in order to assess relative complexity.
4. Calculate the performance of competing systems to determine overall stage weights.
5. Discuss factors relating to secondary propulsion mode selection, based on results of performance and complexity analysis.

2.2 PROPULSION SYSTEM REQUIREMENTS

The primary and secondary propulsion systems must satisfy the requirements shown in Table 10. The main engine provides 8,000 lb of thrust for the orbit insertion burn in all cases, and may or may not be throttleable to 1,000-lb thrust for the secondary ΔV requirements. The throttleable main engine was described during Phase I of this contract. If a new secondary system is provided, it is assumed to be pressure-fed and to have two gimbaling, radiation-cooled engines providing 100-lb thrust each.

Ref. 2 Voyager Support Study, Final Report, Vol. I, Propulsion Studies, TRW Systems Report No. 04480-6004-R000, dated Feb 1967

Table 10

PROPULSION REQUIREMENTS

BURN NO.	TIME AFTER LAUNCH (DAYS)	ΔV (FT/SEC)	APPROXIMATE SPACECRAFT WEIGHT (LB)	MAX IMPULSE LB-SEC
1	3	164	16, 300	83, 000
2	165	164	16, 030	82, 800
3	195	6, 294	15, 760	3, 075, 000
4	205	328	4, 800	49, 000

MINIMUM IMPULSE BIT $\simeq$ 1500 LBF-SEC.
IMPULSE VARIATION LIMIT $\simeq$ $\pm$150 LBF-SEC.

2.3 PRESSURIZATION AND THERMODYNAMIC EFFECTS ON PRIMARY PROPULSION

The objective of the pressurization and thermodynamic analysis was to compare the system weights of a vehicle using the main engine for all burns with those for a vehicle in which a secondary propulsion system is used for all burns except orbit injection

Two sets of computer runs were made, one for a new baseline multiburn main engine condition and one for the single-burn main engine. The thermal-pressurization program used was revised and updated since Phase I and now includes new properties data for hydrogen as developed by the National Bureau of Standards. The ullage was assumed to contain no noncondensable helium prior to the orbit injection burn for the single-burn analysis, thus reducing the optimum total tank pressure significantly.

The results of the thermal/pressurization analyses are shown in Table 11. The major weight reduction for the single-burn system is in propellant vapor and insulation. In each case the weight variables are optimized to give the lowest total combined system weight.

2.4 PROPULSION SYSTEMS DESCRIPTIONS

2.4.1 Propulsion System Description – Mode A, Throttled Main Engine

The systems described provide complete propulsive capability for all required spacecraft ΔV corrections and orbit injection. This is accomplished by full-thrust operation or throttled low-thrust operation as required. A common feed system and pressurization system is utilized by both propulsive modes. These systems incorporate long space coast capability and the simplification necessary for reliable operation.

2.4.1.1 Earth Storables – Mode A. The schematic of the multi-burn earth storable primary system is shown in Fig. 22. This system utilizes two oxidizer tanks and two fuel tanks. Both fuel tanks are manifolded together with a common feed line and common vent line. The oxidizer system employs common manifolds also. No pre-valves

Table 11

PRIMARY SYSTEM WEIGHTS FOR SINGLE VS MULTIBURN

Propellant	Burn	Max. Press. (PSIA)	Insul. Thick (In.)	Weight per Tank*: W_{VAPOR} (LB)	Weight per Tank*: $W_{PRESS.}$ (LB)	Weight per Tank*: $W_{INSUL.}$ (LB)	Weight per Tank*: W_{TANKS} (LB)	Weight per Tank*: ΣW (LB)	Weight All Tanks: ΔW Multi Less Single (LB)
F_2	SINGLE	51	0.75	1.1	7.4	13.2	71.0	92.7	100.3
	MULTI	55	1.25	17.1	5.5	22.5	74.4	119.5	
H_2	SINGLE	68	3.0	7.2	–	144.5	127.2	278.8	
	MULTI	53	4.0	21.5	–	180.5	123.5	325.5	
FLOX	SINGLE	140	0.75	1.4	31.6	11.8	75.2	120.1	83.8
	MULTI	53	2.0	19.3	2.9	38.8	76.3	137.3	
CH_4	SINGLE	127	0.63	0.8	15.0	8.8	66.3	90.9	
	MULTI	191	1.63	18.7	6.1	22.9	67.9	115.6	
N_2O_4	SINGLE	18	0.25	0.2	2.7	6.7	72.3	81.9	14.2
	MULTI	23	0.25	1.7	2.8	6.9	74.1	85.5	
A-50	SINGLE	15	0.25	0	0.8	5.9	69.0	75.7	
	MULTI	15	0.25	0.1	0.8	6.1	72.2	79.2	

*ONE TANK FOR H_2, TWO TANKS FOR ALL OTHER PROPELLANTS are used. Tanks with common liquids utilize a parallel vent system with one relief valve. There is no backup relief because a vent is not anticipated during the mission. If a vent is required, it will be as a result of a system malfunction; therefore, in order for the tanks to rupture due to overpressurization, the system relief valve would have to malfunction. This would be a double failure mode. The relief valve can be opened on the ground from an external facility source. In flight this valve may not be commanded opened. No fill valves are necessary for the main propellants. The function of the fill has been replaced by a check quick disconnect which can be made to seal as well as a poppet valve. The checking function of the disconnect is opened as long as it is mated with the facility. A limit switch may be incorporated to ensure that the check has seated properly. Propellant orientation devices will not have to be employed in any of these tanks. The main engine will start in an idle mode. This low thrust start transient period will provide settling forces that will position the propellants in the sumps of the tanks. Upon complete settling and filling of the feed lines with liquid, the engine will come up to full thrust. The flow of propellants is initiated simply by opening the engine valves.

The pressurization system consists of ambient helium stored in the high-pressure bottle, and the necessary plumbing. The pressurization plumbing features bottle, relief valve, fill system, and regulation system. Helium is filled through a quick disconnect and is closed off with a squib fired valve. In the event of any possible overpressurization within the bottle, a burst disc will rupture and relief valve U-1 will open. The bottle storage pressure is approximately 4,000 psia. The safety valve U-3 is utilized for ground operations and inflight safety. The high pressure in the bottle is initially stepped down by regulator RG-1 to 500 psia. This allows a relatively constant flow condition through the engine heat exchanger. It also provides a regulated tapoff pressure for a pneumatic supply. Pressurization valve U-6 controls the flow and is used as a pressurization shutoff valve. The 500-psi pressure is then regulated to the propellant tanks by two individual regulators RG-2 and RG-3. This provides the proper pressure for each oxidizer tank and fuel tank. Only one regulator is needed for the fuel system and one for the oxidizer system.

All three regulators in the pressurization system will have to have the capability to regulate pressure under widely varying flow conditions. This is due to throttled engine operation. When the low-thrust operation is necessary, the pressurization flow rates will vary nearly linearly with thrust. The variation of flow rate imposes design constraints upon the engine heat exchanger. This engine heat exchanger is not modulated, however, it has the capability for providing the same outlet temperature under widely varying flow conditions. This is accomplished by mixing the hot flow through the heat exchanger with a calibrated bypass loop. The heat exchanger is sized to always bring the hot side gas up to the same temperature, i.e., nearly 100 percent efficient through this loop. The cold bypass temperature is relatively constant at equilibrium temperature. The mixture of the hot gas and cold bypass loop will produce the desired pressurization temperature. Since the ratio of the hot side to the bypass loop will remain nearly constant with varying pressurization requirement, the final temperature will remain nearly constant.

2.4.1.2 <u>Sequence of Operations — Earth Storables — Mode A</u>

I. Purge and Checkout.
   A. Pressurization System.
      (1) Purge bottles through fill disconnect QD2 (Fig. 22) (repeated helium injection and blowdown).
      (2) With bottles slightly pressurized, open pressurization valve U3, oxidizer tank vent U4, and fuel tank vent U5, and pressurization valve U6. Helium will purge all pressurization lines.
      (3) Close pressurization valve U3 and U6.
   B. Oxidizer Tanks and feedline.
      (1) Open fill-and-drain valve at facility.
      (2) Purge tanks with nitrogen through QD8. Gases exit through vent valve U4 that was previously opened.
      (3) While tanks are being purged, open the oxidizer engine valve. The feedline will then be purged and the engine valve may be closed. If the engine valves are hermetically sealed, a purge valve immediately upstream of the engine valve will be opened to allow purging of the feedline.
      (4) Close vent valve U4 and fill valve.

C. Fuel tanks and Feedline.

(1) Open fill-and-drain valve at facility.

(2) Purge tanks with nitrogen through QD3. Gases exit through vent valve U5 that was previously opened.

(3) While tanks are purged, open the fuel engine valve. The fuel feedline will then be purged through the engine valve or alternate purge valve. The engine valve may then be closed.

(4) Close the vent valve U5 and engine valve.

II. Fill

A. Fuel tanks.

(1) Open fuel vent U5 and facility fill valve immediately upstream at the fill discount QD3.

(2) Fill tanks through fuel fill-and-drain quick disconnect QD3. Use slow fill, fast fill and top-off rates as necessary. Both tanks fill simultaneously.

(3) Close facility fill valve immediately upstream of disconnect.

(4) Close vent valve U5 just prior to lift-off.

B. Oxidizer tanks.

(1) Open vent valve U4 and facility fill valve immediately upstream of the fill disconnect QD8.

(2) Fill tanks through oxidizer fill-and-drain quick disconnect QD8.

(3) Close facility fill valve.

(4) Close vent valve U4 just prior to lift off.

C. Pressurization.

(1) Pre-conditioned gaseous helium is introduced at the helium fill disconnect QD2.

(2) After the proper mass has been loaded, the helium squib shut-off valve U2 is fired closed.

III. Flight Operation

A. Startup.

(1) Open pressurization Valve U3.

(2) Purge engine by opening engine purge valve.

(3) Upon engine start signal open engine valves and pressurization valve U6.

B. Shutdown.

(1) Upon engine shutdown command, close engine valves.

(2) Close pressurization valve U6 simultaneously.

(3) Close engine purge valve.

2.4.1.3 Space Storables – Mode A. A space-storable system is similar to the earth-storable system as can be seen in Fig. 24. A checking quick disconnect has replaced the function of a fill valve. The propellants in these lines, however, are subject to heat absorption through thermal radiation. This energy will, in turn, be transferred to the main tank system and impose a weight penalty. However, the penalty associated will be offset by elimination of pre-valves, fill valves, and additional vent valves. Also, the system parts count has been considerably reduced and the overall reliability increased. The pressurization system for the space storables differs from that of the earth storables, mainly in the manner in which the helium is stored. The helium bottles are assumed placed within the oxidizer tanks in order to reduce the gas storage temperature and thus reduce the weight of the bottle. The same effect could be achieved by encapsulating the helium bottle within the insulation of the propellant tank and providing a heat short between the tanks. The helium supply has been divided into two equal bottles in order to keep the volume of the oxidizer propellant tanks equal. Although more heat has to be added to the helium to bring it up to the proper expulsion temperatures, very little additional heat exchanger weight is necessary. Since the main engine will start in an idle mode, no pressurization is necessary and therefore preheating of the pressurant prior to engine operation is not necessary.

2.4.1.4 Sequence of Operations – Space Storables – Mode A

I. Purge and Checkout.

A. Pressurization System.

(1) Purge bottles through fill disconnect QD2 (Fig. 24) (repeated helium injection and blowdown).

(2) With bottles slightly pressurized, open pressurization valve U3, oxidizer tank vent U4, fuel tank vent U5, and pressurization valve U17. Helium will purge all pressurization lines.

(3) Close pressurization valves U3 and U17.

B. Oxidizer Tanks and Feedline.

(1) Open fill-and-drain valve at facility.

(2) Purge tanks with nitrogen through QD8. Gases exit through vent valve U4 that was previously opened.

(3) While tanks are being purged, open the oxidizer engine valve. The feedline will then be purged and the engine valve may be closed. If the engine valves are hermetically sealed, a purge valve immediately upstream of the engine valve will be opened to allow purging of the feedline.

(4) Close vent valve U4 and fill valve.

C. Fuel Tanks and Feedline.

(1) Open fill-and-drain valve at facility.

(2) Purge tanks with nitrogen through QD3. Gases exit through vent valve U5 that was previously opened.

(3) While tanks are purged, open the fuel engine valve. The fuel feedline will then be purged through the engine valve or alternate purge valve. The engine valve may then be closed.

(4) Close the vent valve U5 and engine valve.

II. Fill

A. Fuel Tanks.

(1) Open fuel vent U5 and facility fill valve immediately upstream at the fill disconnect QD3.

(2) Fill tanks through fuel fill-and-drain quick disconnect QD3. (Prechill may be necessary). Use slow fill, fast fill and top-off rates as necessary. Both tanks fill simultaneously.

(3) Close facility fill valve immediately upstream of disconnect.

(4) Close vent valve U5 just prior to lift-off.

B. Oxidizer Tanks.

(1) Open vent valve U4 and facility fill valve immediately upstream of the fill disconnect QD8.

(2) Fill tanks through oxidizer fill-and-drain quick disconnect QD8.

(3) Close facility fill valve.

(4) Close vent valve U4 just prior to lift-off.

C. Pressurization.

(1) Pre-conditioned gaseous helium is introduced at the helium fill disconnect QD2.

(2) After the proper mass has been loaded, the helium squib shut-off valve U2 is fired closed.

III. Flight Operation

A. Startup.

(1) Open pressurization valve U3.

(2) Purge engine by opening engine purge valve.

(3) Upon engine start signal open engine valves and pressurization valve U17.

B. Shutdown.

(1) Upon engine shutdown command, close engine valves.

(2) Close pressurization valve U17 simultaneously.

(3) Close engine purge valve.

2.4.1.5 <u>Cryogens – Mode A</u>. The cryogenic system is considerably different from either earth storables or space storables and special design features are necessary, as illustrated in Fig. 26. Because of the large volume of hydrogen required for the mission, one ellipsoidal hydrogen tank is used. Two oxidizer tanks are used. A pre-valve, fill valve, and vent relief valve are used for the hydrogen tank only. These are necessary on this system because of the more severe penalty associated with heat inputs to the main tank from the feed, fill, and vent lines. The added weight of these components has been offset by the reduction of trapped liquid in the lines. There will be trapped fluid between the pre-valve and engine valve and this amount will vaporize. A check valve has been placed in parallel with the pre-valve to permit the trapped fluid to bleed back into the main tank, relieving pressure within the feed line. The pressurization system consists of a helium supply for the oxidizer tanks and hydrogen bleed tapoff gas from the main engine for the hydrogen tank. A high-pressure helium bottle is stored within the liquid hydrogen tank, resulting in a considerable weight savings.

An engine heat exchanger is used to heat the helium from the liquid hydrogen temperature to the liquid expulsion temperature. As with the other systems, the high-pressure is initially staged down with a regulator upstream of the heat exchanger. A second regulator downstream of the heat exchanger drops the pressure to that required for the oxidizer tank pressurization.

The procedure for filling of the oxidizer tanks is the same as for the space-storable oxidizer tanks. The tanks may be capped off or left vented depending upon facility requirements. In the vented condition, the fluorine gas will have to be disposed of through the appropriate facility disposal units. If this is not permitted because of hazardous conditions, the vent valve will be closed after final tapping off, and a ground thermal conditioning system incorporated into the insulation system.

After filling the hydrogen tank the helium bottle may be loaded. It is necessary that liquid hydrogen be loaded prior to filling of the helium bottle in order to ensure that the bottle and its contents will remain at liquid hydrogen temperatures and prevent overpressurization of the bottle. A squib shutoff valve should not be fired closed until just prior to liftoff so that the helium bottle may be dumped at any time in the event an abort is required.

As was true for the earth storable and space storable systems, there is no difference in operation of the vehicle propellant feed systems between full thrust and throttled operation. No additional plumbing components are necessary for either mode of operation.

2.4.1.6 Sequence of Operations – Cryogens – Mode A

I. Purge and Checkout

A. Pressurization System.

(1) Purge bottles through fill disconnect QD2 (Fig. 26)(repeated helium injections and blowdown).

(2) With bottles slightly pressurized, open pressurization valve U3, oxidizer tank vent U6 and pressurization valve U8. Helium will purge all oxidizer pressurization lines.

(3) Close pressurization valves U3 and U8.

B. Oxidizer Tanks and Feedline

(1) Open fill-and drain valve at facility.

(2) Purge tanks with nitrogen through QD8. Gases exit through vent valve U6 that was previously opened.

(3) While tanks are being purged open the oxidizer engine valve. The feedline will then be purged and the engine valve may be closed. If the engine valves are hermetically sealed, a purge valve immediately upstream of the engine valve will be opened to allow purging at the feedline.

(4) Close vent valve U6 and fill valve.

C. Fuel Tank, Feedline and Pressurization Line.

(1) Open fill-and-drain valve U4, tank shutoff valve U7, and vent valve U5.

(2) Purge tank with nitrogen through QD3. Gases exit through vent valve U5.

(3) While tanks are being purged,open the fuel engine valve. The fuel feed line and pressurization line will then be purged. The tank shutoff valve U7 and engine valve may then be closed.

(4) Close the vent valve U5 and fill-and-drain valve U4.

(5) Repeat steps (1) through (4) with hydrogen gas.

II. <u>Fill</u>

A. Fuel Tanks.

(1) Open fuel vent U5, fill-and-drain valve U4 and facility fill valve immediately upstream of the fill disconnect QD3.

(2) Fill tank through fuel fill-and-drain valve U4 (prechill may be necessary). Use slow fill, fast fill, and top-off rates as necessary.

(3) Close fill-and-drain valve U4.

(4) Close vent valve U5 and facility valve just prior to liftoff.

B. Oxidizer Tanks.

(1) Open oxidizer vent valve U6, and facility fill valve upstream of the fill disconnect QD8.

(2) Fill tanks through oxidizer fill-and-drain quick disconnect QD8. Use slow fill, fast fill, and top-off rates as necessary.

(3) Close facility fill.

(4) Close vent valve U6 just prior to liftoff.

C. Pressurization

(1) Pre-conditioned gaseous helium is introduced at the helium fill disconnect QD2.

(2) After the proper mass has been loaded, the helium squib shutoff valve U2 is fired closed (should be fired just prior to liftoff).

III. Flight Operation

A. Startup

(1) Open pressurization valves U3 and U8.

(2) Purge engine by opening engine purge valve.

(3) Upon engine start signal open tank shutoff valve U7.

(4) Open engine valves.

B. Shutdown.

(1) Upon engine shutdown command, close engine valves.

(2) Close pressurization valve U8 simultaneously.

(3) Close tank shutoff valve U7.

(4) Close engine purge valves.

(5) Open H_2 engine valve only.

C. Restart

(1) Close H_2 engine valve.

(2) Repeat above procedure.

2.4.2 Primary Propulsion System Description – Single Burn – Modes B, C, and D.

The simplified systems for a single-burn primary engine are described in this section. This primary system is used for Propulsion Modes B, C, and D. The second part of this section describes secondary systems with separate tanks.

2.4.2.1 Single-Burn N_2O_4/A-50 Primary Description. The main propellant feed system for a multiburn earth storage system is inherently simple and therefore only modest further simplification is possible. A diagram describing the single-burn system is presented in Fig. 27.

Hermetically sealed engine valves can be installed to eliminate internal leakage prior to engine burn. This will, however, complicate vehicle purging and checkout on the ground. These seals are mechanically ruptured by valve actuation. It is anticipated that leakage through the engine valves would be negligible, however, because liquid will be present at the valve seat. Weight reductions were made in the pressurization system by changing to a flowdown mode as used on the Agena vehicle. Flow control is unmodulated. An orifice calibrates the flow to the fuel and oxidizer tanks. All of the regulators were therefore removed. A burst disc is added to the pressurization shutoff valve to prevent the possibility of helium leakage overpressurizing the tanks, a condition which could otherwise occur because no propellant is removed from the tank for 195 days in Modes B and C. The ullage is quite small during this period and even small amounts of leakage would increase the tank pressure significantly.

Two isolation check valves, CK3 and CK4, were added to the fuel and oxidizer pressurization lines. These replace the shutoff capability of the regulators and help prevent diffusion of the propellants upstream to a common point. A burst disc could be used to replace check valves except that isolation must be maintained after engine burn.

Operation is not unlike that of the multiple-burn pressurization system. The tanks need not be pre-pressurized as does the Agena because the engine is assumed to start in the idle mode, i.e., zero NPSH. Propellant containment devices will aid in engine start. The tank pressure will not be regulated and deviations from optimum will result. The maximum pressure will be held to the lowest possible, while the shutdown pressure will be at or near the minimum premissible consistent with engine requirements.

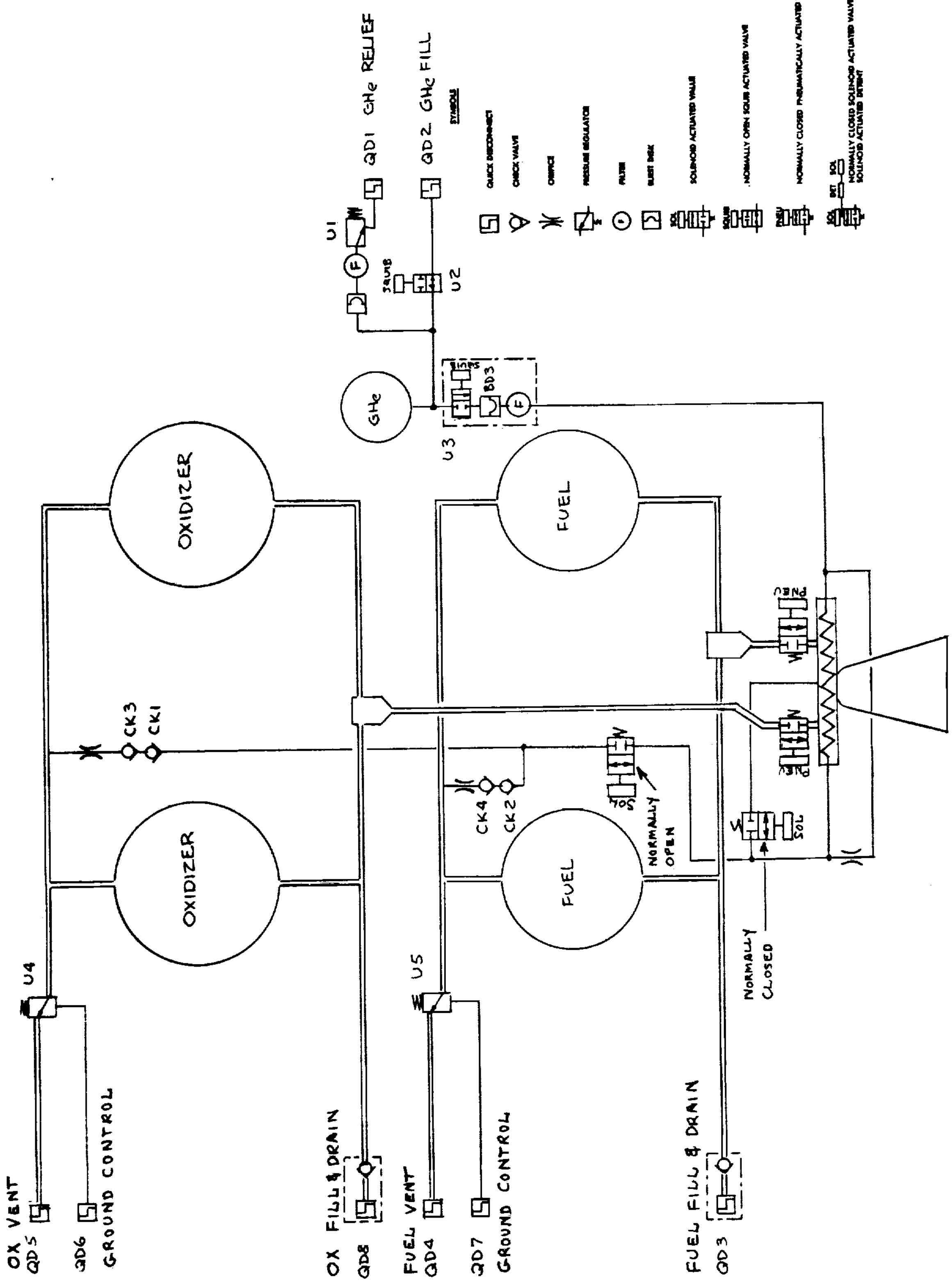


Fig. 27 Mars Orbiter Fluid Systems Schematic – Earth Storable Propellant – Single Burn of Main Engine

The weight saving for a single-burn primary earth-storable system through elimination of components is estimated to be 8.8 lb as follows:

	Component Eliminated	Weight (lb)
RG1	GHe Regulator	2.5
RG2	Oxidizer Pressure Regulator	3.0
RG3	Fuel Pressure Regulator	3.0
U6	Pressurization Valve	1.5
	Weight of Components Eliminated	10.0
	Component Added	
BD3	GHe Burst Disc and Filter	0.2
CK3	Oxidizer Isolation Valve and Orifice	0.5
CK4	Fuel Isolation Valve and Orifice	0.5
	Weight of Components Added	1.2
Net Weight Reduction		8.8

2.4.2.2 Sequence of Operations – Single-Burn N_2O_4/A-50 Primary, Modes B, C, and D

I. PROPELLANT CHECKOUT

A. Pressurization System

1. Purge the helium bottle through quick disconnect QD2 (Fig. 27). The bottle is pressurized and blown down a sufficient number of times to completely purge the bottle of hazardous condensables.

2. The pressurization line downstream of pressurization valve U3 is purged during installation. Helium is trapped between check valves CK2 and CK3 at all times prior to flight. In the event that this method proves unreliable, a purge valve will be installed downstream of pressurization valve U3.

B. Oxidizer Tanks and Feed Lines

1. Open oxidizer vent valve U4 with ground control at QD6.

2. Helium is made available to oxidizer fill and drain disconnects QD8. Open facility valve downstream of disconnect QD8 and purge fill lines, feed lines, both oxidizer tanks, and vent lines. Purge gases will exit through oxidizer vent disconnect QD5. Sweeping action of purged gases will purge the oxidizer feed line between the plenum and engine valve.

3. Close facility valve downstream of oxidizer fill and drain disconnect QD8 and close U4. The entire oxidizer system is now purged and ready for filling.

C. Fuel Tanks and Feed Lines

1. Open fuel vent valve U5 with ground control at QD7.

2. Helium is made available to fuel fill-and-drain disconnects QD3. Open facility valve downstream of disconnect QD3 and purge fill lines, feed lines, both fuel tanks, and vent lines. Purge gases will exit through fuel vent disconnect QD4. Sweeping action of purged gases will purge the fuel feed line between the plenum and engine valve.

3. Close vent valve U3 and facility valve.

II. FILL

A. Fuel Tanks

1. Open fuel vent valve U5 and facility vent valve immediately downstream at fill disconnect QD3.

2. Fill tanks through fuel fill and drain quick disconnect QD3. Use slow-fill, fast-fill, and top-off rates as necessary. Both tanks fill simultaneously.

3. Close facility fill valve immediately upstream of disconnect QD3.

4. Close fill valve U5 just prior to liftoff.

B. Oxidizer Tanks

1. Open vent valves U4 and facility valve immediately upstream of the fill disconnect QD8.

2. Fill tanks through oxidizer fill-and-drain quick disconnect QD8.
3. Close facility fill valve.
4. Close vent valve U4 just prior to liftoff.

C. Pressurization

1. Preconditioned gaseous helium is introduced in the helium fill disconnect QD2.
2. After the proper mass has been loaded, the helium quib shutoff valve is fired closed.

III. FLIGHT OPERATIONS

A. Startup

1. Purge the engine by opening engine purge valves.
2. Open the engine valves
3. At the proper moment relative to engine start signal, fire open squib valve U3. Pressure from the helium bottle will rupture burst disc BD-3 (part of valve U3). It may or may not be necessary to open U3 prior to the engine valves. This is dependent upon tank pressure transient.

B. Shutdown

1. Upon engine shutdown command, close engine valves.
2. Close engine purge valves.
3. Residual helium may overpressurize the propellant tanks. In this case, the vent valves will open as necessary to relieve tank pressure.

2.4.2.3 <u>Single-Burn Space Storable Primary Description</u>

A diagram describing a single-burn space-storable primary system is presented in Fig. 28. This system is quite similar to the earth-storable system, assuming that the space-storable system, including engine, is shaded from the sun. The helium pressurant bottles are now buried within the oxidizer tanks to take advantage of low-temperature storage and reduce bottle size and mass.

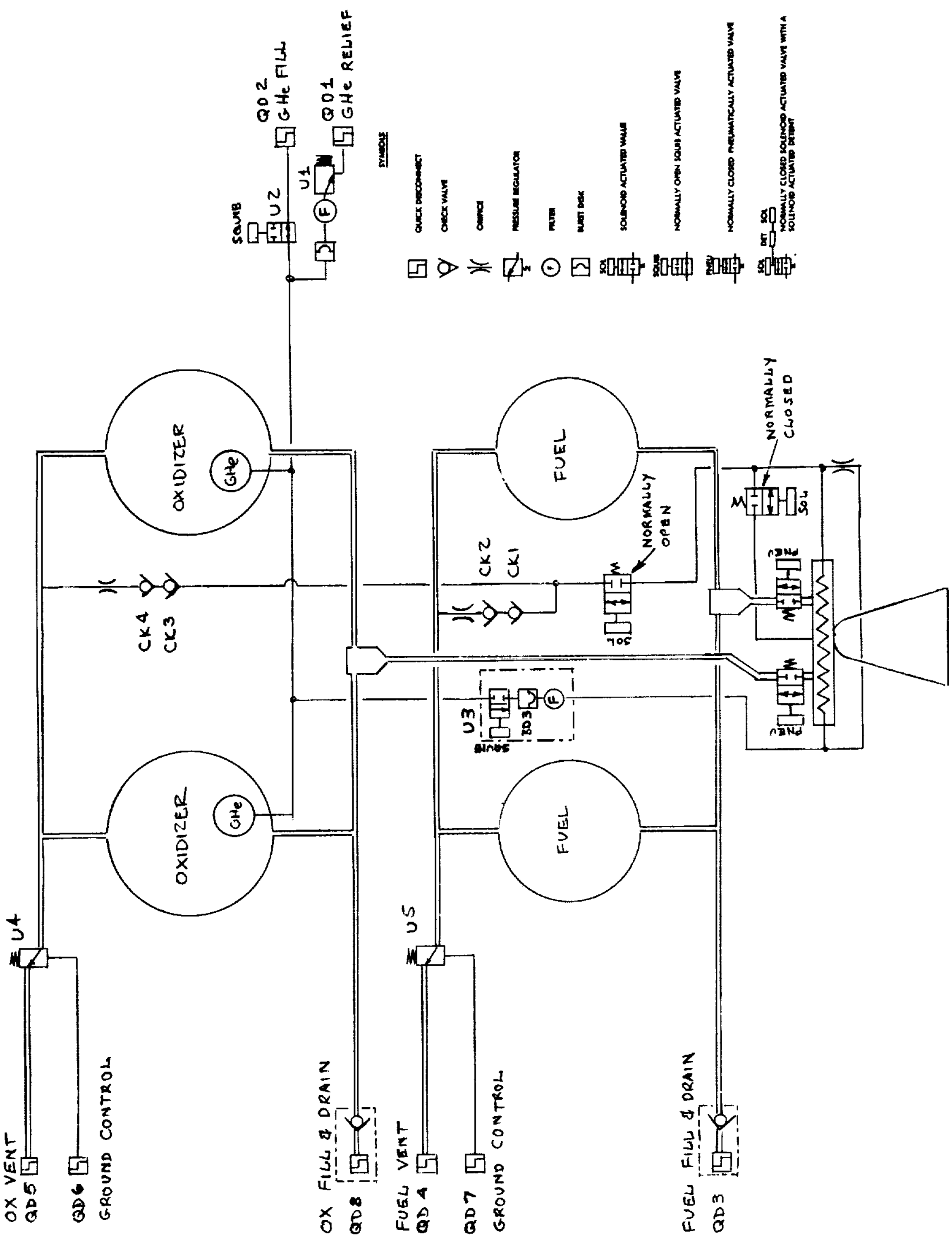


Fig. 28 Mars Orbiter Fluid Systems Schematic – Single Burn of Main Engine

The weight saving for a single-burn primary space-storable system through elimination of components is estimated to be 8.6 lb as follows:

	Component Eliminated	Weight (lb)
RG1	GHe Regulator	2.5
RG2	Oxidizer Pressure Regulator	3.0
RG3	Fuel Pressure Regulator	3.0
U17	Pressurization Valve	1.5
	Weight of Components Eliminated	10.0
	Component Added	
CK3	Fuel Isolation Check Valve and Orifice	0.6
CK4	Oxidizer Isolation Check Valve and Orifice	0.6
BD3	GHe Start Burst Disc and Filter	0.2
	Weight of Components Added	1.4
Net Weight Reduction		8.6

2.4.2.4 Sequence of Operations – Single-Burn Space-Storage Primary, Modes B, C, and D

The sequence of operations for a single-burn space-storable primary is identical with that of the earth storables single-burn primary with one exception. Upon engine start, it may be necessary to open the fuel valve prior to opening of the oxidizer valve. This will establish liquid flow and prevent a hard start.

2.4.2.5 Single-Burn Cryogenic Primary Descriptions

A diagram describing a single-burn cryogenic primary system is presented in Fig. 29. For this system a prevalve at the hydrogen tank is assumed required, and the hydrogen feed line is evacuated until time for engine firing. The hydrogen tank is pressurized by hydrogen bleed gas from the engine. The helium for pressurizing the fluorine tanks is stored in a bottle buried in the hydrogen tank.

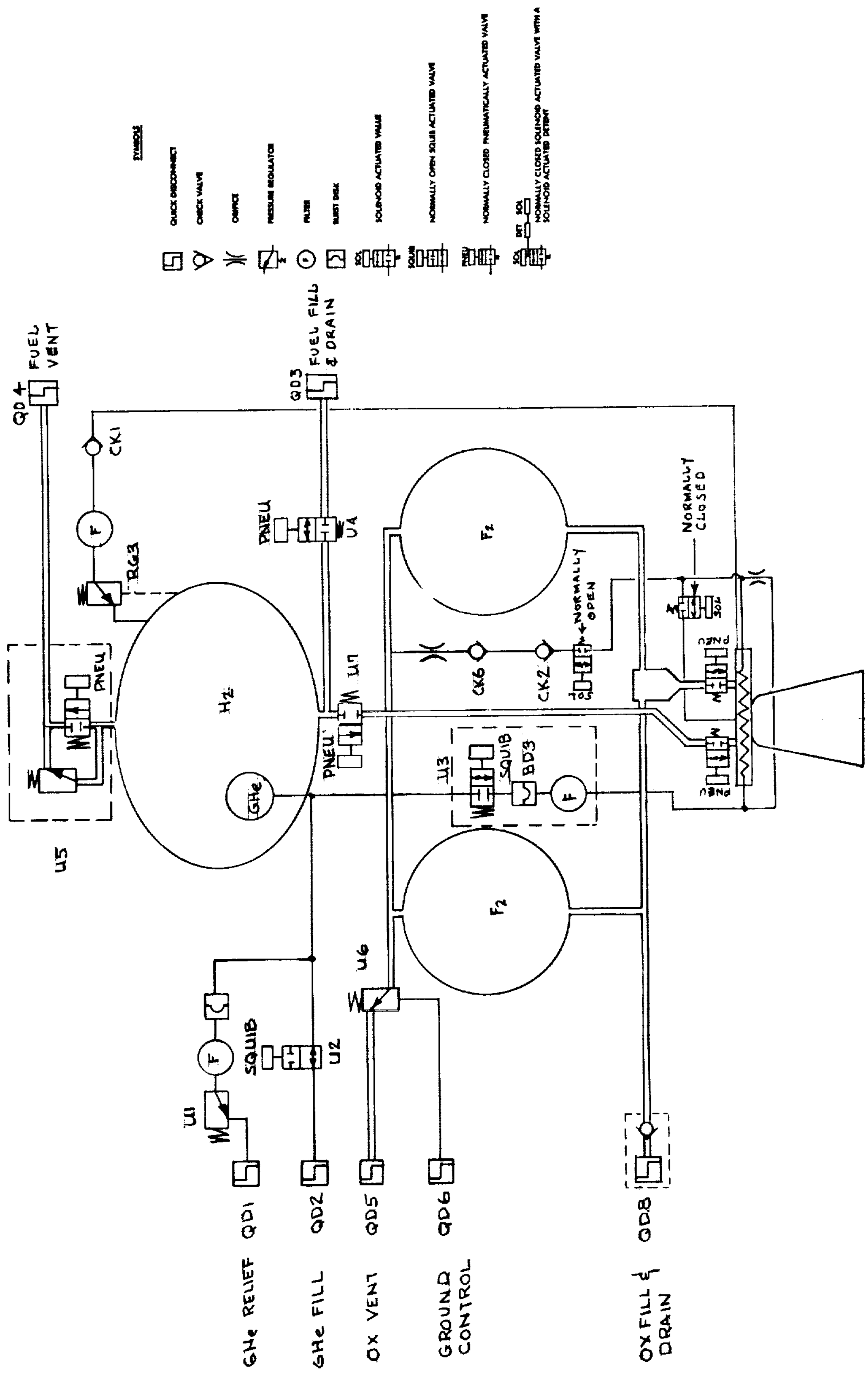


Fig. 29 Mars Orbiter Fluid Systems Schematic – Cryogenic Propellants – Single Burn of Main Engine

The weight saving for a single-burn primary cryogenic system through elimination of components is estimated to be 8.2 lb as follows:

	Component Eliminated	Weight (lb)
CK4	Fuel Thermal Relief	0.5
U8	Oxidizer Pre. Solenoid	3.0
RG1	GHe Regulator	2.5
RG2	Oxygen Tank Regulator	3.0
	Weight of Components Eliminated	9.0
	Component Added	
CK6	F_2 Isolation Check Valve and Orifice	0.6
BD3	GHe Start Burst Disc and Filter (Part of U3)	0.2
	Weight of Components Added	0.8
Net Weight Reduction		8.2

2.4.2.6 Sequence of Operations—Single-Burn Cryogenic Primary, Modes B, C, and D

I. PURGE AND CHECKOUT

A. Pressurization System

1. The helium bottle is purged through fill quick disconnect QD2 (Fig. 29). The bottle is pressurized and blown down a sufficient number of times to completely purge the bottle of hazardous condensables.

2. The pressurization line downstream of pressurization valve U3 is purged during installation in the vehicle. Helium will be trapped between check valve CK6 and the pressurization valve U3 at all times prior to flight. Also, prior purging of the hydrogen bleed pressurization lines is accomplished during installation. If this method proves unreliable, purge valves may be installed on these lines.

B. Oxidizer Tanks and Feed Lines

1. Open facility fill and drain valve downstream of the oxidizer fill and drain disconnect QD8.

2. Open vent valve U6 with ground control.

3. Purge the tanks through disconnect QD8. Gases will exit through the oxidizer vent valve U6. The fill and drain lines, oxidizer tanks, feed lines, and vent lines will be purged simultaneously. The feed line will be purged by sweeping action of the purged gases.

4. After sufficient volumes have been expended, close vent valve U6 and the facility purge valve downstream of disconnect QD8.

C. Fuel Tanks and Feed Lines

1. Open fill and drain valve U4, tank shutoff valve U7, and vent valve U5.

2. Purge the tank through quick disconnect QD3. Gases exit through vent valve U5. Sweeping action of purged gases will purge the fuel feed line. If insufficient sweeping action takes place because of restriction of the fuel prevalve U7, purging of this line may be accomplished during installation.

3. Close vent valve U5, vent and drain valve U4, and engine prevalve U7. Steps 1 through 3 are first completed with nitrogen gas. Steps 1 through 3 are then repeated with hydrogen gas.

II. FILL

A. Fuel Tanks

1. Open vent valve U5, fill and drain valve U4, and facility fill valve immediately upstream of fuel disconnect QD3.

2. Fill tank through fuel fill-and-drain valve U4 (pre-chill may be necessary). Use slow-fill, fast-fill, and top-off rates, as necessary.

3. Close fill-and-drain valve U4.

4. Close fill valve U5 and facility valve just prior to liftoff.

B. Oxidizer Tanks

1. Open oxidizer vent valve U6 and the facility fill valve upstream of disconnect QD8.

2. Fill tank through oxidizer fill-and-drain QD8. Use slow-fill, fast-fill, and top-off rates, as necessary.

3. Close facility fuel valve downstream of QD8.

4. Close vent valve U6 just prior to liftoff.

C. Pressurization

1. Preconditioning gaseous helium is introduced at the helium fill disconnect QD2.

2. After the proper mass has been loaded, the facility shutoff valve is closed. The helium squib shutoff valve U2 is fired closed just prior to liftoff.

III. FLIGHT OPERATIONS

A. Startup

1. Upon initiation of start sequence, pressurization squib valve U3 is fired opened. Helium bottled pressure will rupture burst disc BD3 (part of pressurization valve U3).

   The on-board timer will signal the engine valves and hydrogen prevalves U7 to open. The timing is dependent upon tank pressure transient. Opening of the propellant valves may lead or lag opening of pressurization valve U3.

2. Purge engine by opening engine purge valves.

3. It may be necessary to provide a fuel lead in order to prevent a hard start since vapor will be trapped between the prevalve U7 and engine valve.

B. Shutdown

1. Upon engine shutdown command, close engine valves.

2. Close engine purge valves.

2.4.3 Secondary Systems Description – Mode B – Common Propellant, Separate Tanks

2.4.3.1 N_2O_4/A-50 Secondary With Separate Tanks. A secondary system using N_2O_4/A-50 in separate tanks is used in combination with an N_2O_4/A-50 primary in Mode B and in combination with earth-storable, space-storable, and cryogenic primary systems in Mode D. The fluid systems schematic is shown in Fig. 30.

This system is similar to the primary earth-storable system except that it is patterned after the smaller attitude control feed systems. Both tanks are pressurized with ambient helium stored in a separate bottle. A regulator is used to provide proper tank pressure during expulsion. A surface tension containment device is placed in the bottom of each tank. Enough propellant is trapped to allow engine start and settling of the propellants in the sump region. These devices have been reliably demonstrated through several active flight programs.

By elimination of bladders and associated hardware, a significant savings can be made for tanks of this size. Expulsion bladders normally retard the diffusion of propellant vapors upstream in the pressurization line to a common point. A check valve is normally employed in each pressurization line to further inhibit this diffusion. Since bladders are not being used for this system, series check valves have been installed to take the place of the bladder and single check valve. In order to preclude the possibility of a check valve failing closed, a quad-redundant check valve assembly has been installed in each line as shown in the schematic (CK1 and CK2). If any difficulty is encountered with this pressurization concept, the heavier bladder system may be substituted.

Relief valves are installed on each pressurization line with a burst disc upstream. This will prevent overpressurization of the propellant tanks in the event of a regulator failure or tank overpressure for any other reason. The burst disc in series with the relief valve eliminates leakage which over a 205-day mission can become significant. No fill, pre-valves, or vent valves are necessary for this system. Checking quick disconnects take the place of both fill and vent valves as was done with the main propulsion system. Both propellant tanks are loaded by opening and closing of ground

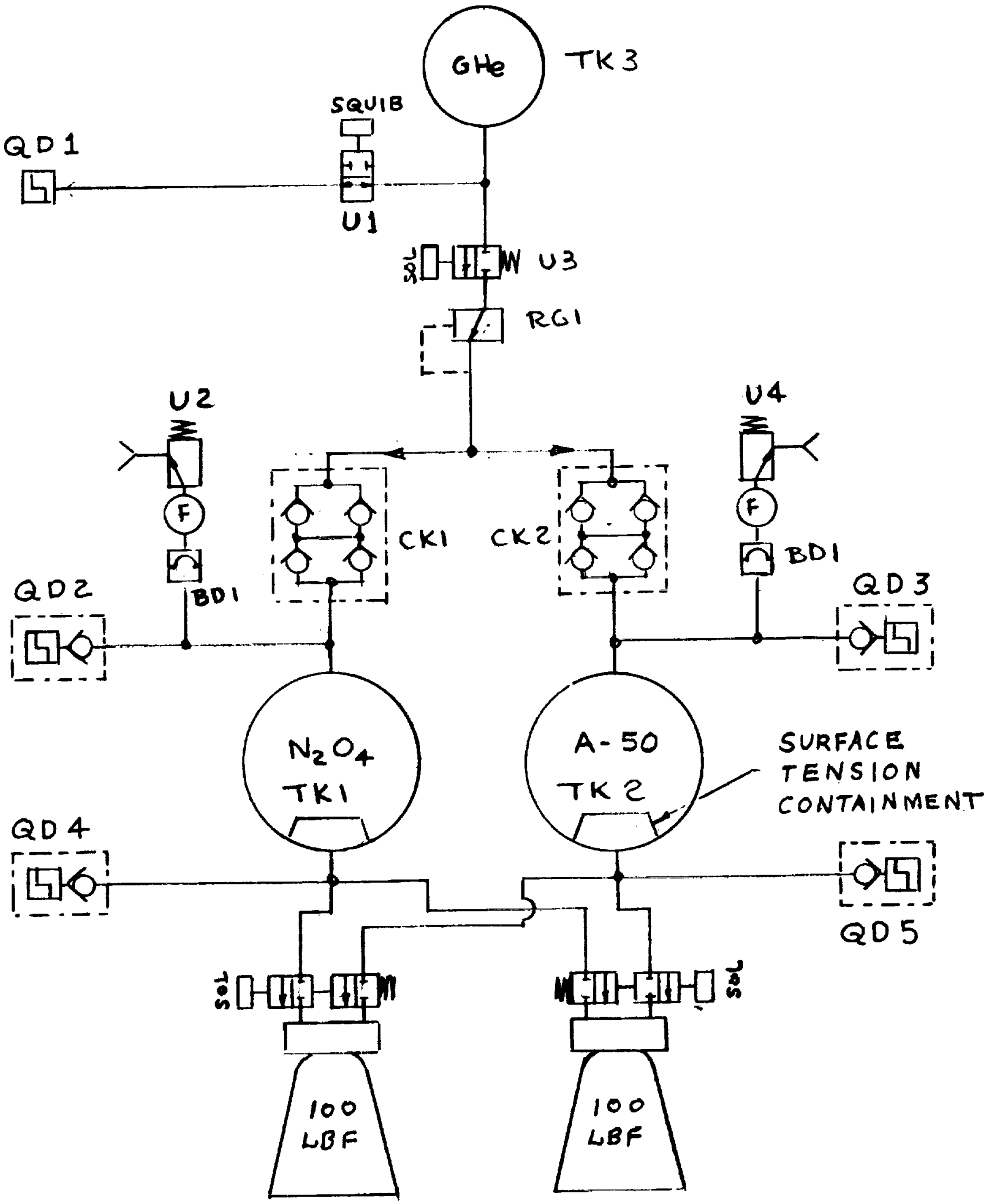


Fig. 30 Mars Orbiter Secondary Propulsion System Schematic-Separate Tankage N_2O_4/A-50

facility valves. The checking fill and vent disconnects are opened when mated to the facility half. These disconnects may be broken just after loading.

Pressure-fed thrusters are utilized for the propulsion units. These units are 100 lb thrust, radiation cooled chambers. Flight qualified articles have been produced. Some minor redesign or modification may be required for extended burn durations.

A list of components, number of components used, and weights for a separate secondary propulsion system using N_2O_4/A-50 is presented in Table 12. Weights of propellant tanks, helium and helium tank, and expulsion bladders are expressed as a fixed weight plus a weight determined by the design propellant loading, i.e., from Table 12,

$$W_{System_{(2)}} = 55 + 0.064\, W_{Prop_{(2)}} + W_{Prop_{(2)}}$$

where

Subscript (2) means secondary system and

Prop = propellant

2.4.3.2 The sequence of operations for an earth-storable secondary system with separate tanks is as follows:

I. PURGE & CHECKOUT

A. Pressurization System

1. The helium bottle is purged through quick disconnect QD-1 (Fig. 30). The bottle is pressurized and blown down a sufficient number of times to adequately purge the bottle.
2. Open pressurization valve U-3 and facility valves downstream of vent disconnects QD-2 and QD-3.
3. Slightly pressurize the helium bottle to purge out all pressurization lines and components. Purging of the propellant tanks is initiated by opening facility valves downstream of fill disconnects QD-4 and QD-5 prior to completion of pressurization purge. Gas will then purge both propellant tanks simultaneously and exit through QD-4 and QD-5. The

Table 12

WEIGHTS OF SECONDARY PROPULSION SYSTEM N_2O_4/A-50 SECONDARY WITH SEPARATE TANKS

Component		Weight (lb)
U1	GH_e Fill Valve	0.3
U2	N_2O_4 Relief Valve	0.6
U3	Pressurization Valve	0.5
U4	A-50 Relief Valve	0.6
BD1	N_2O_4 Relief Burst Disc	0.2
BD2	A-50 Relief Burst Disc	0.2
RG1	Pressurization Regulator	1.9
CK1	N_2O_4 Isolation Check Valve Assembly	1.0
CK2	A-50 Isolation Check Valve Assembly	1.0
QD1	GH_e Quick Disconnect	0.2
QD2	N_2O_4 Checking Vent Disconnect	0.3
QD3	A-50 Checking Vent Disconnect	0.3
QD4	N_2O_4 Checking Fill Disconnect	0.6
QD5	A-50 Checking Fill Disconnect	0.6
TVC	Actuators	14.8
Propellant Flex Line		2.4
Thrust Chambers (2 ea)		10.2
Propellant Tank	N_2O_4	$.017\ W_{prop(2)}$
(Plus Containment Device)	A-50	$.014\ W_{prop(2)}$
GH_e		$.003\ W_{prop(2)}$
GH_e Bottle		$.030\ W_{prop(2)}$
Lines and Fittings		3.0
Tank Supports		3.0
Insulation		3.3
Miscellaneous (Electrical Harness, Supports etc.)		5.0

$$W_{system(2)} = 55 + 0.064\ W_{prop(2)} + W_{prop(2)}$$

fill valves downstream of the vent disconnects can be closed just after initiating tank purge. The pressurization system is now completely purged.

B. Propellant Tank and Feed Lines

1. Open facility valves downstream of QD4 and QD5 prior to completion of pressurization purge as described above.
2. After sufficient volumes have been expended the tanks will be purged of hazardous condensibles. Open the engine valves to purge the feed lines after the tanks are purged.
3. Facility valves downstream of fill disconnects QD-4 and QD-5 and engine are closed after the feedlines are completely purged.

II. FILL

A. Pressurization Systems

1. Solenoid U-3 is closed prior to filling of the pressurization bottle. Ambient helium is introduced at helium fill disconnect QD-1.
2. After proper pressure has been obtained in the helium bottle the supply is shut-off at the facility valve downstream of QD-1.
3. Helium fill valve U-1 is not activated until immediately prior to lift off. This will allow the bottle to be dumped at any time prior to lift off since this is a squib valve. This system is now ready for operation.

B. Oxidizer Tank Fill

1. Nitrogen Tetroxide is introduced to fill disconnect QD-4. Simultaneously a facility fill valve downstream of oxidizer vent disconnect QD-2 is opened to allow ullage gases and vapors to be removed.
2. Filling is initiated with slow fill, then proceeds to fill and is completed with fine fill.
3. After the proper fill level has been attained the facility fill valve is closed.

C. Fuel Tank

1. Aerozine-50 is introduced to fill disconnect QD-5.
2. Simultaneously the fuel tank facility vent valve downstream of QD-3 is opened. The fill procedure is the same as for Oxidizer Tank.
3. After the proper fill level has been obtained the facility fill valve and vent valves are closed.

III. FLIGHT OPERATIONS

A. Start Up

1. Prepressurization is accomplished by opening pressurization valve U-3.
2. After the proper pressure has been obtained the engine valves are opened. The engines will now come up to full thrust.

B. Shut Down

1. Upon engine shut down command the engine valves are closed.
2. Simultaneously close pressurization valve U-3.

2.4.3.3 Space-Storable Secondary With Separate Tanks

A secondary system using FLOX/CH_4 in separate tanks is used in combination with a FLOX/CH_4 primary in Mode B. The fluid systems schematic is shown in Fig. 31. This system is quite similar to the earth-storable system except that certain measures are taken to provide for operation of the cold propellants.

Transfer valves and fill valves have been added to replace the function of the quick disconnects used in the earth storable system. This was done to ease design requirements for a fluorine compatible checking quick disconnect. A slight weight penalty will be paid by the addition of these valves but will be considered negligible.

Burst disc and relief valve assembly has been added to the pressurization system to protect the helium bottle from overpressure. This is necessary in the event that the helium bottle is subjected to ambient temperatures on the ground after the bottle has been charged. Since the helium is stored cold it is necessary to be heated prior to injection in order to reduce the mass of helium required for expulsion. Small engine heat exchangers were added to provide this heating function.

The space storable system is closely integrated to the main propulsion system for thermal protection. The helium bottle and secondary system oxidizer tank may be enclosed within the main system oxidizer tank insulation. Also a thermal heat short between the secondary and main tank will be installed in order to insure proper thermal

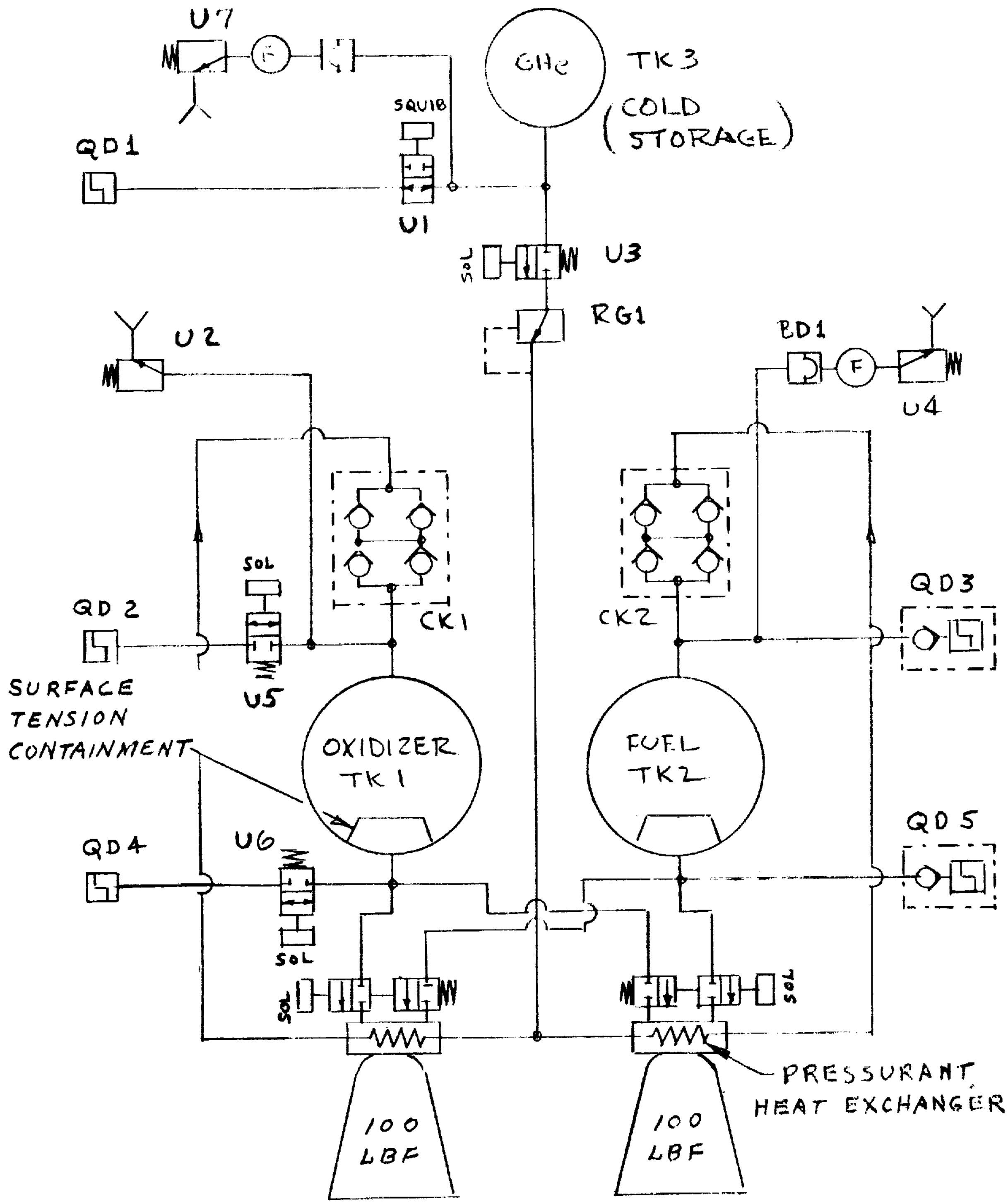


Fig. 31 Mars Orbiter Secondary Propulsion System Schematic-Separate Tankage FLOX/CH_4 or F_2/H_2

control of the propellant. The fuel tanks of each system will also be thermally linked. The insulation around the secondary system tankage is assumed to be the same thickness as for the main propellant tanks.

A list of components, number of components used, and weights for a separate secondary propulsion system using FLOX/CH_4 is presented in Table 13. Total weight of secondary system is determined by the following expression, derived from Table 13.

$$W_{system(2)} = 55 + 0.060\ W_{prop(2)} + 0.125\ W^{2/3}_{prop(2)} + W_{prop(2)}$$

2.4.3.4 The sequence of Operations for a space-storable secondary system with separate tanks is as follows:

I. Purge and Checkout

A. Pressurization System

1. The helium bottle is purged through quick disconnect QD-1 (Fig. 31). The bottle is pressurized and blown down a sufficient number of times to adequately purge the bottle. A facility valve is used to control the flow of pressurant.
2. Open pressurization solenoid valve U-5 and facility valve downstream of the fuel tank vent quick disconnect QD-3.
3. Slightly pressurize the bottle to purge out all pressurization lines and components. Gas will exit through disconnect QD-2 and QD-3. The purge flow is not terminated until initiation of propellant tank purging.

B. Propellant Tank and Feed Lines

1. Open oxidizer fuel valve U-6, facility valve downstream of fill disconnect QD-5 and all engine valves.
2. Flow of hazardous condensibles and purge gases will now exit through QD-4, QD-5 and the engine valves. Shortly after this initiation the

Table 13

WEIGHTS OF SECONDARY PROPULSION SYSTEM
$FLOX/CH_4$ PRIMARY AND SECONDARY WITH SEPARATE TANKS

	Component	Weight (lb)
U1	GH_c Fill Valve	0.3
U2	FLOX Relief Valve	1.0
U3	Pressurization Valve	0.5
U4	CH_4 Relief Valve	0.6
U5	FLOX Ground Vent Valve	0.8
U6	FLOX Fill Valve	1.0
BD1	CH_4 Relief Burst Disc	0.2
RG1	Pressurization Regulator	1.9
CK1	FLOX Isolation Check Valve Assembly	1.5
CK2	CH_4 Isolation Check Valve Assembly	1.0
QD1	GH_e Quick Disconnect	0.2
QD2	FLOX Quick Disconnect	0.2
QD3	CH_4 Checking Vent Disconnect	0.3
QD4	FLOX Fill Quick Disconnect	0.4
QD5	CH_4 Fill Checking Disconnect	0.6
TVC	Actuators	14.8
Propellant Flex Line		2.4
Thrust Chambers (2 ea)		10.2
Tank Supports		7.0
Engine Heat Exchanger		1.0
Insulation		$.126\ W^{2/3}_{prop}$
U7	GH_e Bottle Relief Valve	0.5
Propellant Tank FLOX		$.022\ W_{prop(2)}$
	CH_4	$.0129\ W_{prop(2)}$
GH_e		$.0051\ W_{prop(2)}$
GH_e	Bottle	$.0204\ W_{prop(2)}$
Lines & Fittings		3.0
Miscellaneous (Electrical Harness, Supports, etc.)		5.0

$$W_{system} = 55 + 0.060\ W_{prop(2)} + 0.126 W^{2/3}_{prop(2)} + W_{prop(2)}$$

propellant feed lines will be completely purged and may be closed. If sufficient volumes have been expended through the main tanks fill valve U-6 and the facility valve downstream of QD-5 will be closed. The tank fill lines and feed lines are now completely purged and ready for filling.

II. FILL

A. Pressurization Systems

1. Solenoid U-3 is closed prior to filling of the pressurization bottle. Helium is introduced at helium fill disconnect QD-1.
2. After proper pressure has been attained in the Helium bottle the supply is shut off at the facility valve downstream of QD-1.
3. Helium fill Valve U-1 is not activated until immediately prior to lift off. This will allow the bottle to be dumped at any time prior to lift off since this is a squib valve. The system is now ready for operation.

B. Oxidizer Tank Fill

1. FLOX is introduced to fill disconnect QD-4. Fill valve U-6 and fill valve U-5 are opened simultaneously to fill the oxidizer tank.
2. Filling procedure is initiated with slow fill, then proceeds to fast fill and is completed with fine fill.
3. After the proper fill level has been attained fill valve U-6 is closed. The vent valve may be closed at this time if a ground hold vent-free conditioning system is incorporated into the secondary propulsion tanks. This will eliminate the need to dispose of hazardous fluorine gas. If this vent valve is left open and the oxidizer tank is not thermally conditioned the fluorine gases will have to be disposed of through proper facility disposal units.

C. Fuel Tank

1. Liquid Methane is introduced to fill quick disconnect QD-5.
2. A facility valve downstream of vent disconnect ED-3 is opened to allow gases to escape.
3. The fuel tank is filled with the proper slow fill, and fine fill.

4. After the proper level has been attained the facility fuel valve is closed. Since Methane is less hazardous than the Fluorine vapors, this tank may be left vented during ground hold and the vapors disposed of by burning.

III. FLIGHT OPERATIONS

A. Start Up

1. Pressurization is accomplished by opening pressurization valve U-3.
2. After the proper pressure has been attained the engine valves are opened. The engine will now come up to full thrust.

B. Shut Down

1. Upon engine shut down command the engine valves are closed.
2. Simultaneously close pressurization valve U-3.

2.4.3.5 Cryogenic Secondary With Separate Tanks

A secondary system using F_2/H_2 in separate tanks is used in combination with a F_2/H_2 primary is Mode B. The fluid systems schematic is shown in Fig. 31. This system is essentially the same as the space-storable except that the sizes of components on the hydrogen side must be increased since the volumetric flow rates are greater than for liquid methane. The fuel tank is also considerably larger, and a greater supply of helium for pressurization is required. The secondary system hydrogen tank is assumed to be buried within the primary hydrogen tank for thermal protection. Table 14 presents data for a separate secondary system using F_2/H_2 where total weight of secondary system is determined by the following expression, derived from Table 14,

$$W_{system(2)} = 57 + 0.086\, W_{prop(2)} + 0.068\, W_{prop(2)}^{2/3}$$

$$+ \left(W_{H_2\ Insul\,(1)} + W_{H_2\ Tank\,(1)}\right)\left[\left(\frac{W_{prop(2)} + W_{prop(1)}}{W_{prop(1)}}\right)^{2/3} - 1\right]$$

$$+ W_{prop(2)}$$

Table 14

WEIGHTS OF SECONDARY PROPULSION SYSTEM
F_2/H_2 Primary and Secondary With Separate Tanks

Component Weights, Same as $FLOX/CH_4$ Plus 2 lb = 57 lb

Insulation, $F_{2(2)} = +0.068\, W^{2/3}_{Prop(2)}$

Secondary H_2 Tank is Buried in Primary H_2 Tank

$$\Delta W_{H_2\,Insulation\,(1)} = W_{H_2\,Insulation\,(1)} \left[\left(\frac{W_{Prop\,(2)} + W_{Prop\,(1)}}{W_{Prop\,(1)}} \right)^{2/3} - 1 \right]$$

$$\Delta W_{H_2\,Tank\,(1)} = W_{H_2\,Tank\,(1)} \left[\left(\frac{W_{Prop\,(2)} + W_{Prop\,(1)}}{W_{Prop\,(1)}} \right)^{2/3} - 1 \right]$$

Secondary Propellant Tanks:

$$F_{2\,Tank\,(2)} = 0.0228\, W_{Prop\,(2)}$$

$$H_{2\,Tank\,(2)} = 0.0395\, W_{Prop\,(2)}$$

He Tank is Buried in H_2 Tank:

$$GHe\,(2) = 0.012\, W_{Prop\,(2)}$$

$$GHe\ Bottle\,(2) = 0.012\, W_{Prop\,(2)}$$

$$W_{System\,(2)} = 57 + 0.086\, W_{Prop\,(2)} + 0.068\, W^{2/3}_{Prop\,(2)} + \left(W_{H_2\,Insulation\,(1)} + W_{H_2\,Tank\,(1)} \right) \left[\left(\frac{W_{Prop\,(2)} + W_{Prop\,(1)}}{W_{Prop\,(1)}} \right)^{2/3} - 1 \right] + W_{Prop\,(2)}$$

where =

subscript (1) refers to primary propulsion system
subscript (2) refers to secondary propulsion system
Insul is insulation
Prop is propellant

The sequence of operations for a cryogenic secondary system with separate tanks is essentially the same as that described for space storables.

2.4.4 Propulsion System Description – Mode C – Common Propellant, Common Tanks

The simplified systems for a single-burn primary engine are described earlier and are essentially identical for Modes B, C. and D. In this section secondary systems operating off propellants transferred from the primary system tanks are described. The secondary thrusters are pressure fed and a pressure transfer system is required. The transfer system proposed consists of feed lines, transfer pumps, a pump drive, and surge accumulators.

2.4.4.1 N_2O_4/A-50 Secondary With Common Tanks

An earth-storable system is shown schematically in Fig. 32. The heart of this system is the direct-drive double-activation pump and the pump drive. Such a pump is presently under development by LMSC and has demonstrated efficient operation with N_2O_4 and A-50. Helium is used as the medium to drive the pumps. This helium would probably be a part of the main system supply, but for weight estimating purposes is assumed here to be separate. The pump has dual-activation bellows, suction and delivery check valves, and a directional control valve. The directional control valve directs activation gas alternately to the two activation chambers in the pump. A continuous flow of propellant is delivered with a minimum of pressure surges. The surges are effectively eliminated by the surge accumulators A1 and A2.

In order to prevent cavitation in the pump a slight pressure in the main tank is necessary at all times. This can be provided by the auxiliary pressurization valve U7.

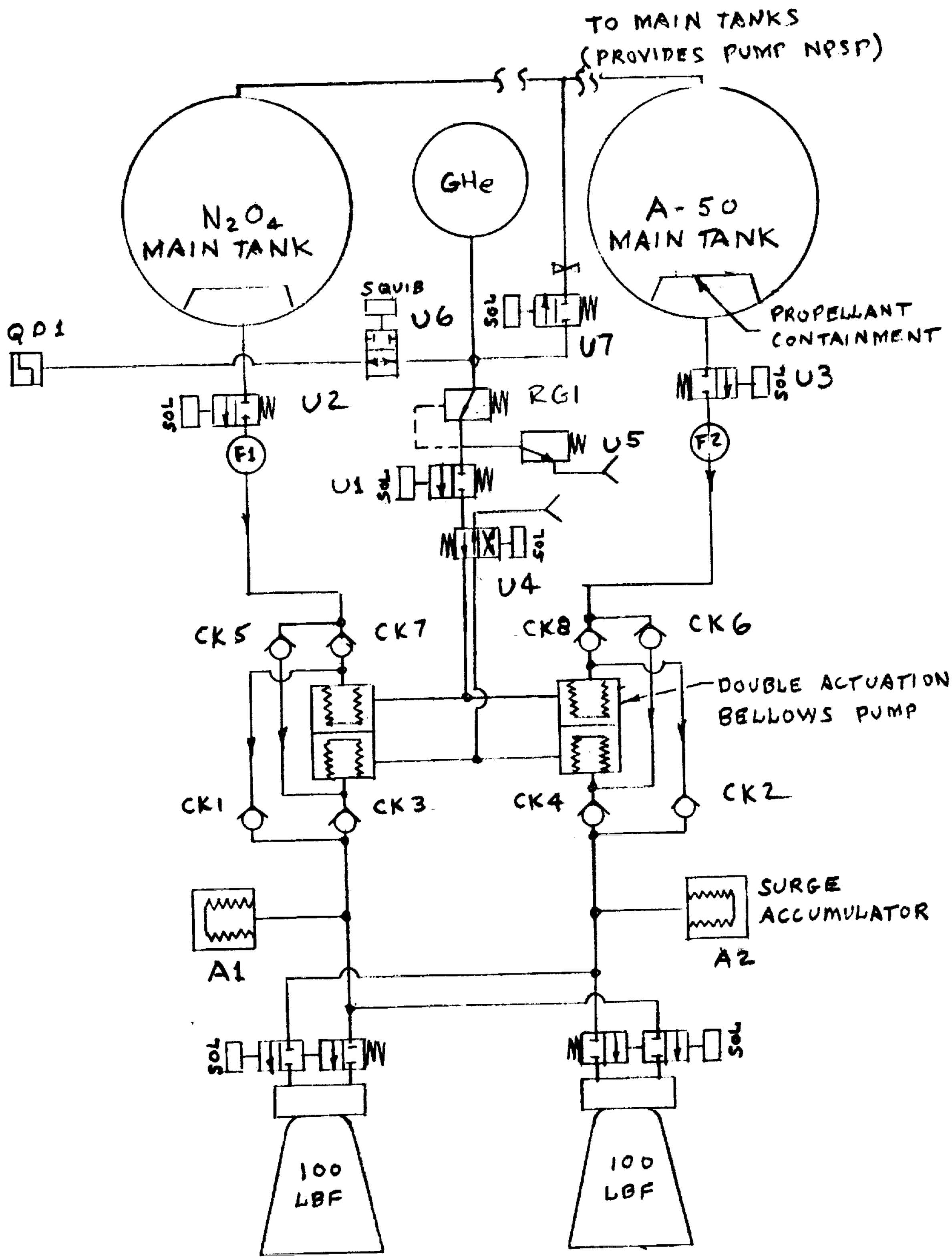


Fig. 32 Mars Orbiter Secondary Propulsion System Schematic Integrated With Main Tanks N_2O_4/A-50

The system need not be primed prior to liftoff with the present design concept. Propellant containment devices have been added to the main tank to insure that ullage gases are not injected into the pump prior to settling of the propellant.

A list of components, number of components used, and weights for an integrated secondary propulsion system using N_2O_4/A-50 is presented in Table 15.

2.4.4.2 The sequence of operations for an earth-storable secondary system with common tanks is as follows:

I. PURGE AND CHECKOUT

A. Pneumatic Supply

1. The helium pneumatic supply bottle will be purged by introducing helium at fill disconnect QD-1. The bottle is pressurized and blown down a sufficient number of times to adequately purge the bottle.
2. Auxiliary prepressurization valve U-7 is opened momentarily to allow purging of the prepressurization line.
3. Pneumatic lines and pump bellow chambers will be purged next. The pneumatic bottle is partially purged, and solenoid start valve U-1 is opened allowing purge gases to actuate the pumps.
4. Control valve U-4 is actuated several times to cycle the pumps. This procedure will drive all gases out of the pneumatic lines and pumps and will be expended.

B. Propellant Transfer Lines.

1. Both propellant tanks are slightly pressurized.
2. Isolation valves U-2 and U-3 and all engine valves are opened.
3. Control valve U-4 is cycled a number of times in order to further actuate both pumps. A decrease in pressure in the suction chamber of each pump will be created by this action. Purge gases will then flow through both propellant lines past check valves CK-1 through CK-8 as the pumps are being cycled. This is repeated until all hazardous condensibles are removed from the system.

Table 15

WEIGHTS OF SECONDARY PROPULSION SYSTEM
N_2O_4/A-50 Secondary With Common Tanks

Component		Weight (lb)
U6	Pneumatic Fill	0.4
U1	Start Valve	1.5
U2	Oxidizer Isolation Valve	1.5
U3	Fuel Isolation Valve	1.5
RG1	Pneumatic Regulator	1.5
A1	Oxidizer Surge Accumulator	1.0
A2	Fuel Surge Accumulator	1.0
CK1, CK3, CK5, CK7	OX Pump Directional Control Check Valves	0.8
CK2, CK4, CK6, CK8,	Fuel Pump Directional Control Check Valves	0.8
F1	Oxidizer Pump Filter	0.7
F2	Fuel Pump Filter	0.7
U4	Pneumatic Control Valve	2.0
U5	Pneumatic Relief Valve	1.0
U7	Pump NPSP Valve	0.5
TVC	Actuators (4)	14.8
Propellant Flex Line (4)		2.4
Fuel Pump		8.0
Oxidizer Pump		8.0
Ports (6)		0.6
Lines & Fittings		2.7
Miscellaneous		7.0
Electrical Control Box & Harness		3.0
GH_e		$.0033\ W_{prop}$
GH_e Bottle		$.033\ W_{prop}$
Thrust Chambers (2)		10.0
Trapped Propellant (3 Burns in Pumps)		3.0
Propellant Containment (Main Tanks)		3.0

$$W_{sys} = 78 + 0.036\ W_{prop(2)} + W_{prop(2)}$$

4. Isolation valves U-2 and U-3, solenoid start valve U-1 and engine shutoff valves are all closed simultaneously. The system is now entirely purged and ready for fill.

II. FILL

A. Pneumatic Bottle

1. High pressure helium is introduced to helium-fill quick-disconnect QD-1.
2. The bottle is purged to the proper pressure.
3. After proper pressure has been obtained and the helium bottle supply is shutoff at the helium valve downstream of QD-1.
4. Helium valve U-6 is not activated until immediately prior to lift off. This will allow the bottle to be dumped at any time prior to lift off since this is a squib valve. The system is now ready for operation.

B. Transfer Line Fill

1. The transfer line will not fill until after launch.

III. FLIGHT OPERATIONS

A. Transfer Line Fill

1. Open all engine valves to space conditions. This will bleed down all propellant feed lines to one or two psia (the cracking differential pressure of the pump flow control check valves CK-1 through CK-8).
2. Close all engine valves after sufficient time has elapsed for bleed off to occur.
3. Open isolation valves U-2 and U-3. After the transfer lines are filled close isolation valves U-2 and U-3. This will fill the lines to approximately 90% volume. The remaining 10% will consist of gaseous helium at the first startup.

B. Start Up

1. Open auxiliary pressurization valves U7 as required and slightly pressurize main fuel and oxidizer tanks. This will provide the necessary NPSH to the transfer pumps to prevent cavitation.
2. Open isolation valves U-2 and U-3.

3. Upon engine start the engine valves are opened and thrust is built up. It may be necessary however to lead with the fuel prior to ignition in order to prevent a hard start.
4. Simultaneously with opening of the engine valves start valve U-1 will be opened.

C. Engine Shutdown
1. Simultaneously start valve U-1 and all engine valves are closed.
2. Close isolation valves U-2 and U-3.

2.4.4.3 $FLOX/CH_4$ Secondary With Common Tanks

A secondary system using $FLOX/CH_4$ transferred from the main tanks of a $FLOX/CH_4$ primary is used in Mode C. The fluid-systems schematic is shown in Fig. 33. This system is basically the same as the earth-storable integrated system shown in Fig. 32. The main changes incorporated are necessary because of the heat absorbed by the propellants during coast and because the helium supply is stored at oxidizer temperature.

A relief valve and burst disc assembly (U-8) are added to prevent overpressurization of the supply valve. Also relief valves (U-6 and U-7) are added to the propellant feed lines to relieve pressures built up due to boil off. The relief gases are directed back to the main tanks. These propellant lines may be completely dried out between engine burns. The pumps are kept separate and are maintained on the tank and under the insulation system. A slight pressurization of the main tank is required to prevent cavitation of the pumps during operation. This pressurization gas results in a slight weight penalty to the main tank pressurization requirements. It is however quite small since during the initial burns of the secondary system the ullage in the main tank is quite small and only a couple of psi NPSH is required. (Approximately 18 pounds of pneumatic bottled weight is saved by storing the helium at oxidizer temperatures and a slight increase in main propellant tank weight is realized.) Since the helium is stored cold it will have to be heated. This is done with a heat exchanger on each engine. Since boil-off occurs in the lines a longer start transient than is normally associated with attitude thrusters will result. Because of the unpredictable volumes associated with this boil off it may be necessary to open the engine fuel valves to provide a fuel lead. This

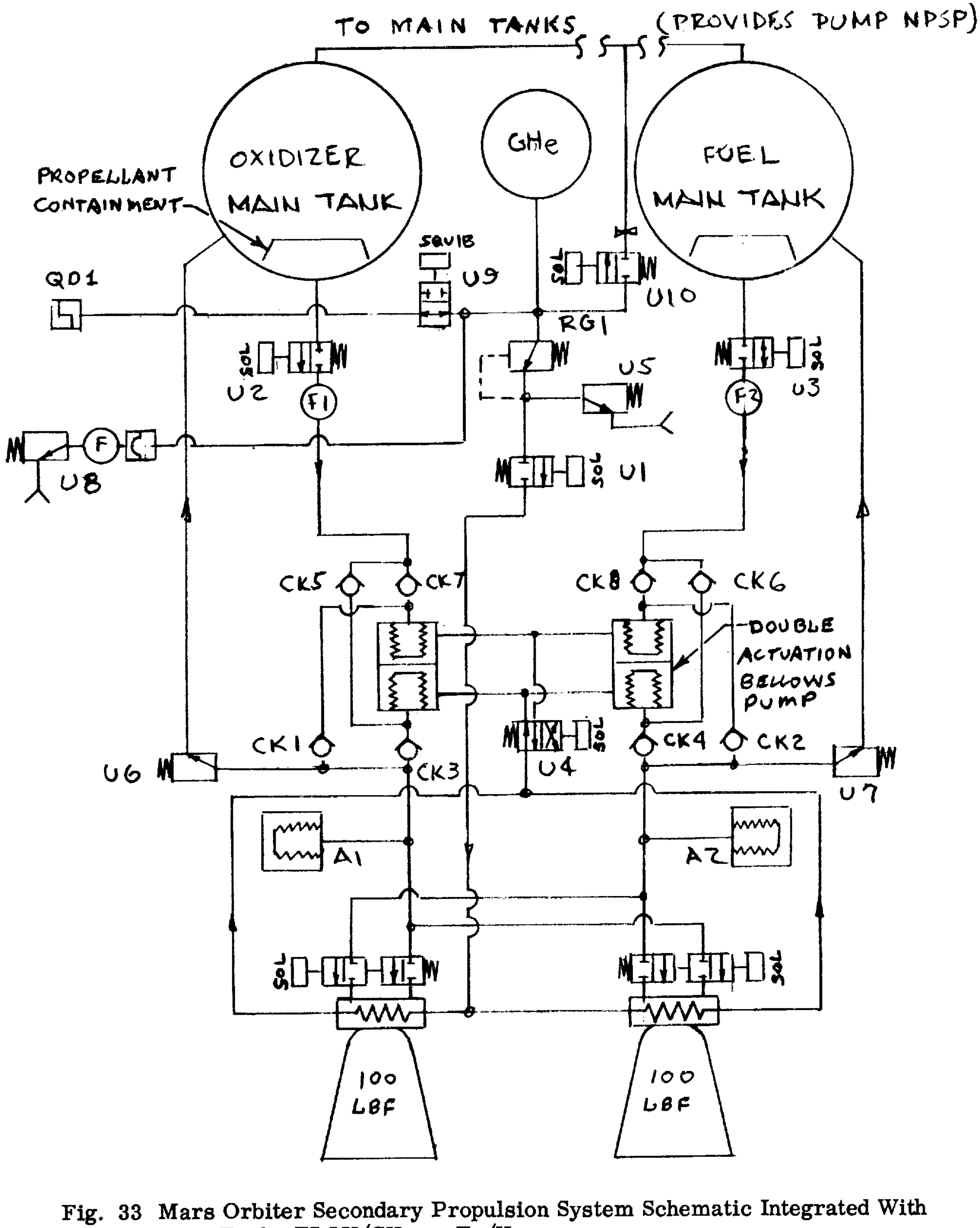


Fig. 33 Mars Orbiter Secondary Propulsion System Schematic Integrated With Main Tanks FLOX/CH_4 or F_2/H_2

will reduce the hazards of a hard start. A list of components, number of components used, and weights for an integrated secondary propulsion system using FLOX/CH_4 is presented in Table 16.

2.4.4.4 Sequence of Operations, FLOX/CH_4 Secondary With Common Tanks

The sequence of operations for both the space storables and cryogens is nearly identical to that of the earth storables. The only potential deviation is during engine start. It may be necessary to eliminate filling of the propellant feed lines prior to opening of the engine valves. If the lines are filled prior to opening of the valves, violent boiling will occur until the lines are chilled down. The boil-off gases may escape through the relief valves U-6 and U-7 but the chilldown may not be sufficient to provide a predictable start. It may be advisable therefore to open the start valve and engine valves simultaneously after having previously evacuated the feed lines as described for the earth-storable system. All other operations including shut down are unaltered.

2.4.4.5 F_2/H_2 Secondary With Common Tanks

A secondary system using F_2/H_2 transferred from the main tanks of a F_2/H_2 primary is also a combination used in Mode C. This system is essentially identical to that of the space-storable system described in Fig. 33. Because of the increased volume on the hydrogen side, the transfer pump and other components are heavier than for liquid methane. These weight differences are reflected in the weight statement presented in Table 17.

The sequence of operations for a cryogenic secondary system with common tanks is identical to that for a space-storable secondary system with common tanks as described earlier.

2.4.5 Propulsion System Description – Mode D – N_2O_4/A-50 Secondary, Separate Tanks

A secondary propulsion system using N_2O_4/A-50 propellants in separate tanks is used in Mode D. The primary system is either earth-storable, space-storable, or cryogenic.

Table 16

WEIGHTS OF SECONDARY PROPULSION SYSTEM
FLOX/CH_4 Secondary With Common Tanks

	Component	Weight (lb)
U1	Start Valve	1.5
U2	Oxidizer Valve	1.5
U3	Fuel Isolation Valve	1.5
U4	Pneumatic Control Valve	2.0
U5	Pneumatic Relief Valve	1.0
U6	Oxidizer Pump Relief Valve	1.0
U7	Fuel Pump Relief Valve	1.0
CK1 →	CK8 Pump Directional Control Check Values	1.6
F1	Oxidizer Pump Filter	0.7
F2	Fuel Pump Filter	0.7
A1	Oxidizer Surge Accumulator	1.0
A2	Fuel Surge Accumulator	1.0
RG1	Pneumatic Regulator	1.5
U10	Pump NPSP Valve	0.5
TVC	Actuators (4)	14.8
Propellant Flex Line (4)		2.4
Fuel Pump		6.0
Oxidizer Pump		10.0
Ports (6)		0.6
Lines & Fittings		3.7
Miscellaneous (Including Insulation)		8.0
Electrical Control Box & Harness		3.0
GH_e		$.0054\ W_{prop(2)}$
GH_e Bottle		$.0216 W_{prop(2)}$
Thrust Chambers (2)		10.0
Trapped Propellant (3 Burns in Pumps)		2.0
Propellant Containment		3.0

$$W_{system} = 81 + 0.027\ W_{prop(2)} + W_{prop(2)}$$

Table 17

WEIGHTS OF SECONDARY PROPULSION SYSTEM
F_2/H_2 Secondary With Common Tanks

	Component	Weight (lb)
U1	Start Valve	1.5
U2	Oxidizer Isolation Valve	1.5
U3	Fuel Isolation Valve	2.0
U4	Pneumatic Control Valve	2.2
U5	Pneumatic Relief Valve	1.2
U6	Oxidizer Pump Relief Valve	1.0
U7	Fuel Pump Relief Valve	1.2
CK1 → CK8	Pump Directional Control Check Valves	1.6
F1	Oxidizer Pump Filter	0.7
F2	Fuel Pump Filter	0.9
A1	Oxidizer Surge Accumulator	1.0
A2	Fuel Surge Accumulator	1.0
RG1	Pneumatic Regulator	1.5
U8	Relief Valve, Burst Disc & Filter Assembly	0.5
U9	Pneumatic Fill Valve	0.4
TVC	Actuators (4)	14.8
Propellant Flex Line (4)		2.4
Fuel Pump		13.0
Oxidizer Pump		9.0
Ports (6)		0.6
Lines and Fittings		4.0
Miscellaneous		9.0
Electrical Control Box & Harness		3.0
GH_e		$.0126\ W_{prop(2)}$
GH_e Bottle		$.0126\ W_{prop(2)}$
Thrust Chambers (2)		10.0
Trapped Propellant (3 Burns in Pumps)		2.0
Propellant Containment		3.5

$$W_{system} = 89 + 0.025\ W_{prop(2)} + W_{prop(2)}$$

Mode D with an N_2O_4/A-50 primary system is identical to Mode B with an N_2O_4/A-50 primary system as described earlier in Figs. 27 and 30.

Mode D with a FLOX/CH_4 primary and an N_2O_4/A-50 secondary is described by referring to the N_2O_4/A-50 secondary system and the FLOX/CH_4 single-burn primary system of Mode B, Figs. 28 and 30.

Mode D with a F_2/H_2 primary and an N_2O_4/A-50 secondary is described by referring to the N_2O_4/A-50 secondary system and the F_2/H_2 single-burn primary system of Mode B, Figs. 29 and 30.

2.4.6 Propulsion Mode E – Idle Mode Secondary

Insufficient data are available at this time to make a quantitative evaluation of use of a primary system in the idle mode for accomplishing secondary ΔV maneuvers. Opinions have been expressed by rocket engine company representatives to the effect that (1) reliability of propellant feed in tank head idle mode for long burn times has not been demonstrated. (2) engine cooling for long burn times in tank head idle mode must also be demonstrated.

The pressurization system requirements would be essentially unchanged, I_{sp}'s would be lowered, and no weight savings realized over Mode A, primary engine throttled.

2.4.7 Propulsion Mode F – ACS Secondary

Propulsion Mode F proposes the use of the attitude control system (ACS) for mid-course and orbit adjust maneuvers. The ACS high thrust level specified by TRW in Ref. 3 totals 12 lb for four thrusters (two pitch and two yaw). If this system were used for a specified 164 ft/sec midcourse maneuver with a 16,000 lb spacecraft, the burn time would be:

Ref. 3 NASA CR71482 TRW 5410-6001-R0V02 Voyager Spacecraft Vol. 2. Preferred Design: Subsystems 17 Jan 1966.

$$\Delta t = \frac{\Delta V}{a} = \frac{164}{\frac{12 \times 32.2}{16000}} = \frac{164}{.0242} \simeq 6800 \text{ sec or,} \quad \Delta t \simeq 113 \text{ minutes}$$

In addition, use of a cold gas system to provide the relatively large total impulse required for secondary maneuvers would add several thousand pounds of mass and an unacceptable volume to the spacecraft.

A low-level ACS thrust of much less than one pound must be provided for the Voyager-class Mars Orbiter mission, and it is through these small thrusters, in limit cycle cruise, that most of the ACS propellant is expended. There is therefore little to be gained by using a different propellant and system for high-level ACS requirements. If larger thrusters were used for high-level-thrust spacecraft orientation maneuvers, and also used for midcourse and orbit trim ΔV corrections, the thrust level would have to be in the order of 20 pounds. This thrust would be marginally high for the high-level ACS requirements and marginally low for midcourse ΔV corrections. The only feasible system using common propellants between fractional-pound thrust low-level and 20 pound thrust high level systems appears to be hydrazine monopropellant, used as a rocket fuel for high-level and as a gas in the low-level system. This might reduce the total ACS system weights by 40 to 50 pounds maximum over a cold N_2 system, but the weights of midcourse and orbit trim will increase total system weight by about 500 pounds over Mode A. Mode F is therefore not recommended.

2.5 SYSTEM PERFORMANCE

System performance is evaluated by comparing the total weight for the baseline stage using a multi-burn main engine with total weights for stages using alternate secondary propulsion systems.

The alternate propulsion modes are described in Table 9. The weight changes for Modes B, C, and D are presented in Tables 18, 19, and 20 respectively. These weight were obtained by:

1. Deleting dry weight savings shown in Table 11 and in paragraphs 2.4.2.1, 2.4.2.3, and 2.4.2.5.

Table 18

PRIMARY VERSUS SECONDARY PROPULSION FOR MINOR ΔV
STAGE WEIGHT CHANGES FROM MODE A TO MODE B
(Secondary System Uses Some Propellants As Primary And Has Separate Engines and Separate Tanks)

Weight Item	F_2/H_2		$FLOX/CH_4$		N_2O_4/A-50	
	Delete Multi-Burn On Primary	Add Secondary	Delete Multi-Burn On Primary	Add Secondary	Delete Multi-Burn On Primary	Add Secondary
Dry Weight	-109	+121	- 93	98	- 41	+105
Secondary ΔV Propellant	-447	+494	-521	+574	-675	+750
Primary ΔV Propellant	—	+ 13	—	+ 10	—	+ 64
Sum	-556	+628	-614	+682	-716	+919
Difference	+72		+68		+203	

Table 19

SECONDARY VERSUS PRIMARY PROPULSION FOR MINOR ΔV
STAGE WEIGHT CHANGES FROM MODE A TO MODE C
(Secondary System Uses Propellants From Main System Tanks And Has Transfer Pumps, Accumulators, and Separate Engines)

Weight Item	F_2/H_2		FLOX/CH_4		N_2O_4/A-50	
	Delete Multi-Burn On Primary	Add Secondary	Delete Multi-Burn On Primary	Add Secondary	Delete Multi-Burn On Primary	Add Secondary
Dry Weight	0	+102	0	+ 97	- 17	+105
Secondary ΔV Propellant	-447	+525	-521	+582	-675	+751
Primary ΔV Propellant	—	+ 70	—	+ 68	—	+ 83
Sum	-447	+697	-521	+747	-692	+939
Difference	+250		+226		+247	

Table 20

PRIMARY VERSUS SECONDARY FOR MINOR ΔV
STAGE WEIGHT CHANGES FROM MODE A TO MODE D
(Secondary System Is N_2O_4/A-50 And Has Separate Engines
And Separate Tanks From Primary Propulsion)

Weight Item	F_2/H_2		FLOX/CH_4		N_2O_4/A-50	
	Delete Multi-Burn On Primary	Add Secondary	Delete Multi-Burn On Primary	Add Secondary	Delete Multi-Burn On Primary	Add Secondary
Dry Weight	- 109	+101	- 93	+102	- 41	+105
Secondary ΔV Propellant	-447	+688	-521	+704	-675	+750
Primary ΔV Propellant	—	+ 26	—	+ 30	—	+ 64
Sum	-556	+815	-614	+836	-716	+919
Difference	+259		+222		+203	

2. Calculating and deleting primary propellant use in Mode A for secondary maneuvers.
3. Calculating weight of propellant required for a new secondary propulsion system.
4. Calculating new secondary system dry weights from appropriate $W_{system(2)}$ equations shown in Tables 12 through 17.
5. Calculating additional primary ΔV propellant required for the orbit insertion burn to account for increased total spacecraft weight over Mode A.
6. Iterating all calculations as necessary to arrive at a balanced system.
7. Summing deletions and additions to find net weight difference for each concept as compared with Mode A.

In each case the lightest weight system is the baseline Mode A wherein the main engine, at full thrust or throttled, is used for all ΔV maneuvers. The next lightest system is mode B using a secondary system with separate engines, separate tanks, and the same propellant as the primary.

2.6 SYSTEMS COMPLEXITY

An attempt is made here to assess the relative complexity of the systems described for propulsion Modes A through D. In this assessment the number of critical components and the number of component critical operational cycles are counted.

A critical component is defined here as one wherein a single failure within that component would cause the mission to be partially or totally lost. Critical components, then, include propellant and pressurant valves, pressure regulators, pumps, and TVC gimbal actuators, but do not include redundant check valves, relief valves, or burst discs. An operational cycle for a component is assumed to consist of once "on" at initiation of a burn and once "off" at the completion of that burn. Cycling during a burn, such as occurs for pumps and pressure regulators, is not counted.

The complexity and desirability of a throttable engine is widely controversial among both engine company engineers and user agency representatives and a reliability comparison of throttable vs non-throttlable engines would be very useful. Under the budget conditions of the contract no funding has been available for engine company support, and though considerable support has been given at no cost, these companies were neither able nor expected to provide the detailed system information necessary to a main engine complexity analysis. For these reasons the complexity of the working parts of the main engine, beyond the fuel and oxidizer valves, was not treated in the analysis, and the only weight adjustment was for ablative material for the N_2O_4/A-50 primary system.

The count of critical components and operations is summarized in Table 21. A ranking of complexity by mode and propellant combination is offered below, assuming, for lack of specific reliability data, that all critical components (1) are equally reliable, (2) have a long mean-time-to-failure compared to required operating time (3) if the number of critical operations is identical, then earth-storables will be considered less complex than space-storables and space-storables less complex than cryogens. The complexity ranking then, based on number of critical operations, is presented in Table 22 with the least complex systems shown first.

A description of each system is presented below. It attempts to point out those considerations, in addition to a count of critical parts and number of operations, which tend to influence the overall reliability of the system.

Propulsion Mode A – Earth-Storables

The earth-storable system described in Fig. 22 has as few parts as any other considered. It does not have the least number of components acutations. Each of the components shown in the propellant feed system is inherently reliable. Since these components are associated with earth-storable propellants the problems of expansion and contraction are less than for space-storables or cryogens.

Table 22
COMPLEXITY RANKING

Complexity Ranking	Propulsion Mode	Primary Propellant	Secondary Propellant
1	B & D	N_2O_4/A-50	N_2O_4/A-50
2	D	FLOX/CH_4	N_2O_4/A-50
3	B	FLOX/CH_4	FLOX/CH_4
4	D	F_2/H_2	N_2O_4/A-50
5	B	F_2/H_2	F_2/H_2
6	A	N_2O_4/A-50	–
7	A	FLOX/CH_4	–
8	A	F_2/H_2	–
9	C	N_2O_4/A-50	N_2O_4/A-50
10	C	FLOX/CH_4	FLOX/CH_4
11	C	F_2/H_2	F_2/H_2

Propulsion Mode A – Space Storables

Comparing the earth-storable system shown in Fig. 22 with the space-storable system in Fig. 24 the systems schematics are nearly identical. The parts count and total number of operations are identical as shown in Table 21. This system is inherently somewhat less reliable than the earth-storable system since fluorine rated components are installed and temperature changes are geater.

Propulsion Mode A – Cryogenics

The cryogenic system shown in Fig. 26 is more complex than either the space storable or earth storable systems. This is because a pre-valve is used for the hydrogen tank and bleed check valves are installed in the hydrogen feed line. The number of engine components should remain about the same. All of the in-flight operating components

Table 21

SUMMARY OF CRITICAL COMPONENTS AND FLIGHT C

Propulsion Mode	Propellants Main + Secondary	Numbers of Individual Critical Components (I			
		Pressurant Valves	Pressure Regulators	Transfer Pumps	Gi Actu
A – Throttleable Main Engine	N_2O_4/A-50 + None	3 + 0	3 + 0	0 + 0	2
	FLOX/CH_4 + None	3 + 0	3 + 0	0 + 0	2
	F_2/H_2 + None	3 + 0	3 + 0	0 + 0	2
B – Single Burn Main Engine Common Propellants Separate SPS	N_2O_4/A-50 + N_2O_4/A-50	3 + 1	0 + 1	0 + 0	2
	FLOX/CH_4 + FLOX/CH_4	3 + 1	0 + 1	0 + 0	2
	F_2/H_2 + F_2/H_2	3 + 1	1 + 1	0 + 0	2
C – Single Burn Main Engine Common Propellants Integrated SPS	N_2O_4/A-50 + N_2O_4/A-50	3 + 2	3 + 1	0 + 2	2
	FLOX/CH_4 + FLOX/CH_4	3 + 2	3 + 1	0 + 2	2
	F_2/H_2 + F_2/H_2	3 + 2	3 + 1	0 + 2	2
D – Single Burn Main Engine Separate N_2O_4/ A-50 SPS	N_2O_4/A-50 + N_2O_4/A-50	3 + 1	0 + 1	0 + 0	2
	FLOX/CH_4 + N_2O_4/A-50	3 + 1	0 + 0	0 + 0	2
	F_2/H_2 + N_2O_4/A-50	3 + 1	1 + 1	0 + 0	2

GHT OPERATIONS

ents (Main + Secondary)		Sum of Critical Components	Number of Critical Operations
Gimbal Actuators	Propellant Valves		
2 + 0	2 + 0	10 + 0 = 10	40
2 + 0	2 + 0	10 + 0 = 10	40
2 + 0	3 + 0	11 + 0 = 11	44
2 + 4	2 + 4	7 + 10 = 17	37
2 + 4	2 + 4	7 + 10 = 17	37
2 + 4	3 + 4	9 + 10 = 19	39
2 + 4	2 + 6	10 + 15 = 25	55
2 + 4	2 + 6	10 + 15 = 25	55
2 + 4	3 + 6	11 + 15 = 26	56
2 + 4	2 + 4	7 + 10 = 17	37
2 + 4	2 + 4	7 + 10 = 17	37
2 + 4	3 + 4	9 + 10 = 19	39

FOLDOUT FRAME 2

will be subjected to cryogenic operation except for a couple of components in the pressurization system. These components are somewhat less reliable than equivalent components for earth-storable systems. The effective temperature will not have a very significant effect upon the overall reliability, and the relative ranking of this system is indicated by the total number of flight operations and parts. This system is therefore somewhat less reliable than either the earth-storable or space-storable systems.

Propulsion Mode B – Earth-Storables

Specifying a single burn for the primary propulsion allows the earth-storable system to be simplified as shown in Fig. 27. The simplication lies mainly in the pressurization system which has been changed to a blowdown mode to eliminate pressure regulators. The total number of critical parts and operations for the single burn configuration as opposed to the multiple burn configuration has been greatly decreased.

The earth-storable secondary system with separate tanks is shown in Fig. 30. The parts count and number of operations on this system is quite low. Only a pressurization start valve and engine valves are necessary to activate the entire system to full thrust. This mode of operation is true generally with most small attitude control systems and has been assumed here in the interest of simplicity and increased reliability. The total number of critical parts for the primary and secondary propulsion systems together has increased by seven over the Mode A earth-storable system, but the number of critical operations has decreased from 40 to 37.

Propulsion Mode B – Space Storables

The single burn space-storable system is simplified as shown in Fig. 28. The comments made concerning the earth-storable Mode B system also apply here since the systems are similar. Low temperature propellants and fluorine compatibility will reduce the overall reliability of the system, but this can only be qualitatively assessed here.

The space-storable secondary system with separate tanks is shown in Fig. 31. The parts and operations count is identical to the earth-storable system but parts are modified somewhat to account for fluorine compatibility and low temperature propellant. This system therefore is considered somewhat less reliable than the equivalent earth storable system.

Propulsion Mode B – Cryogenics

The cryogenic system shown in Fig. 29 has been simplified for single burn operation in the same manner as the earth-storable and space-storable systems. However, only the fluorine pressurization system could be simplified since hydrogen gas bleed from the engine is used for hydrogen tank pressurization. A burst disc has also been added to the hydrogen shutoff valve. The number of critical components and operations is increased by two over the corresponding space-storable primary system.

The cryogenic secondary system with separate tanks is shown in Fig. 31, and is identical to that for the space-storable. The only difference in reliability is due to the extremely low temperature of hydrogen propellants and the associated technical design problems of these components. This system is therefore considered only slightly less reliable than an equivalent space storable system.

Propulsion Mode C – Earth Storable Propellants

The primary system for Mode C is the same as for Mode A operation.

The earth-storable secondary system with integrated propellant tanks is shown in Fig. 32. The parts count for this system is quite a bit greater than for completely separate secondary systems since pump transfer of propellants from the main tanks is utilized.

The overall reliability of Mode C appears to be poorer than for the competing systems.

Propulsion Mode C – Space Storables

The Mode C Space-storable primary system is identical to that for Mode A.

The space-storable secondary with integrated propellant tanks is shown in Fig. 33. The operation is nearly identical to that of the corresponding earth-storable system, with two exceptions. The first is that between burns propellants in the feed lines to the engine will boil out and be released back to the main tanks through relief valves. This will complicate start up procedures somewhat and reduce start reliability. Such conditions can be overcome either by priming of the lines prior to start, by utilizing bleed valves, or by a fuel lead. Completely separate engine valves would then be necessary and would double the number of engine valve solenoids required. The other condition leading to less reliable operation is in the bellows and surge accumulator design. Difficulty has been encountered in obtaining extended life of bellows pumps in cryogen liquids. This problem has been encountered by Bell Aero Systems in the development of a integrated attitude control system for fluorine/hydrogen service. Difficulty was also encountered during the design of a bellows actuated fluorine shutoff valve developed by the J. C. Carter Company. These problems should be overcome in the near future and the reliability of pumping operation increased, but it appears that the reliability of pumps for fluorine service will remain less reliable than for earth-storable propellants.

Propulsion Mode C – Cryogenics

The Mode C cryogenic primary system is identical to that for Mode A.

The cryogenic secondary with integrated propellants is shown in Fig. 33 and is identical to that for the space-storable. The comments as to pump design and control valve operation also apply here. The propellants will be vaporized from the feed lines between burns as was pointed out for the space-storable system. The difference in reliability between the space-storable and cryogenic Mode C systems is equivalent to the difference between the space-storable and cryogenic systems of Mode B, with Mode C inherently less reliable.

Propulsion Mode D – All Propellant Combinations

The primary systems for Mode D are identical to those described earlier for Mode B.

The secondary system for Mode D is the N_2O_4/A-50 system described earlier under Mode B. This secondary system is used regardless of propellant specified for the primary.

This is inherently a reliable combination of systems. One additional precaution must be taken with the systems using space-storable or cryogenic primaries to insure that the secondary earth-storable propellants do not freeze in any part of the system. This may require locating tanks in the warm equipment compartment of the spacecraft and/or providing heat to tanks and feed lines.

2.7 PERFORMANCE AND COMPLEXITY SUMMARY

A summary of propulsion mode analysis results is presented in Table 23, including total propulsion stage weight, weight penalty, and complexity ranking, by mode and primary propellant. When complexity count is identical between two propellant choices, the warmer propellant is assumed to be less complex.

Both weight penalty and complexity count are plotted on Fig. 34. Here it is evident that the lightest weight system is with throttled main engine. Complexity is decreased at a small weight penalty for separate secondary systems with separate propellant tanks. The heaviest and most complex mode is the separate secondary system transferring propellant from the main tanks. N_2O_4/A-50 and FLOX/CH_4 are shown to be of equal complexity, with F_2/H_2 complexity slightly higher.

Table 23

PROPULSION MODE ANALYSIS RESULTS SUMMARY

Propellants	Secondary Propulsion Mode (1)	A	B	C	D (2)	Modes Rejected: E	Modes Rejected: F (3)
N_2O_4/A-50	Stage Weight (lb)	17,678	17,881	17,925	17,881	Feed and Cooling Not Adequately Developed for Long Burn	Cold Gas Thrust & I_{sp} Too Low
	Weight Penalty (lb)	0	203	247	203		
	Complexity	6th	1st (4)	9th	1st (4)		
FLOX/CH_4	Stage Weight (lb)	16,111	16,179	16,337	16,333	Feed and Cooling Not Adequately Developed for Long Burn	Cold Gas Thrust & I_{sp} Too Low
	Weight Penalty (lb)	0	68	226	222		
	Complexity	7th	3rd	10th	2nd		
F_2/H_2	Stage Weight (lb)	15,433	15,505	15,683	15,692	Feed and Cooling Not Adequately Developed for Long Burn	Cold Gas Thrust & I_{sp} Too Low
	Weight Penalty (lb)	0	72	250	259		
	Complexity	8th	5th	11th	4th		

(1) Modes are defined in Table 9
(2) N_2O_4/A-50 Secondary Propellant
(3) ACS Requirements for a Mars Orbiter are easily met with a small, cold gas system
(4) N_2O_4/A-50 Systems for Modes B and D are identical.
(5) Primary is pump fed. Results may differ for a pressure fed primary.

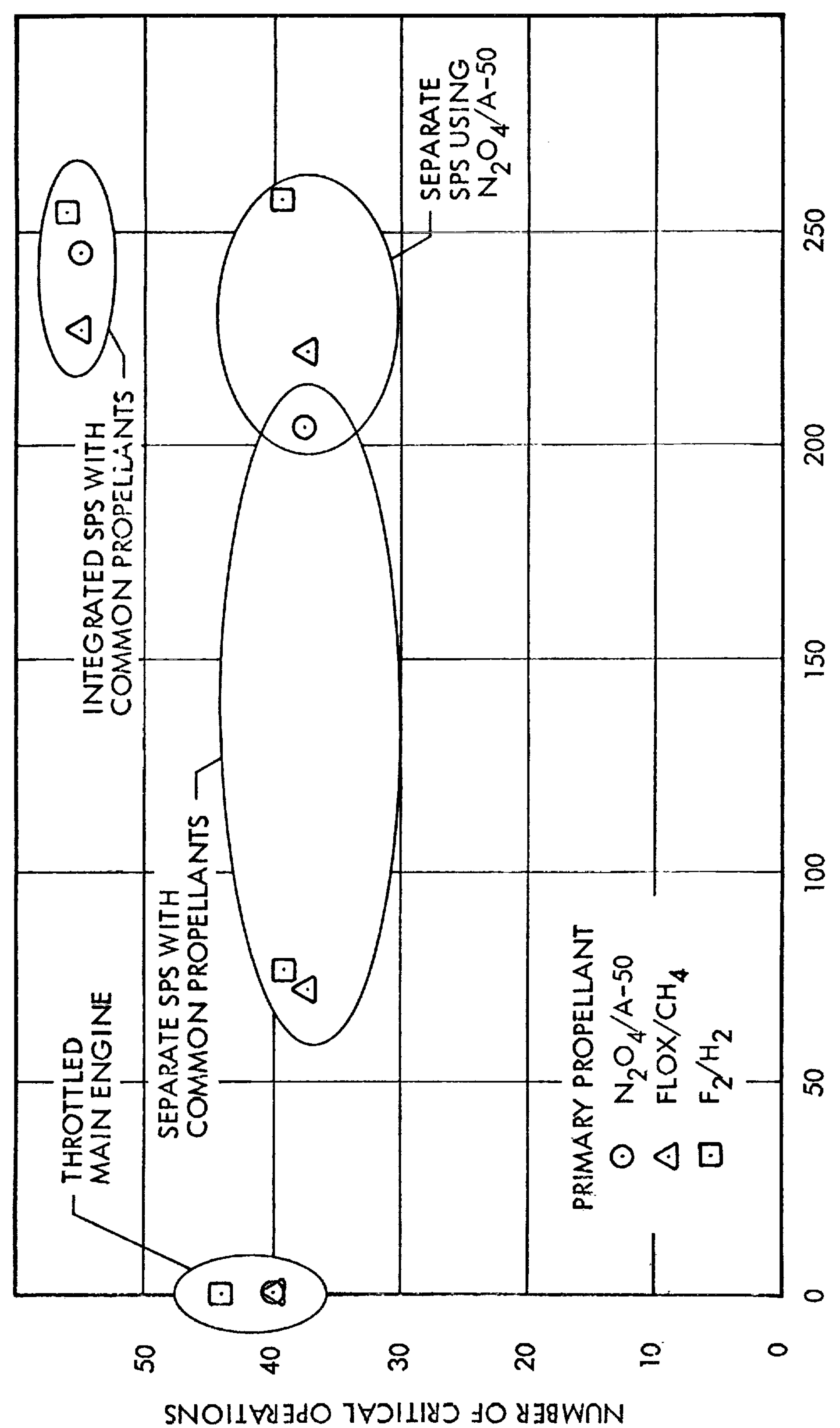


Fig. 34 Secondary Propulsion Complexity and Weight Penalty

Section 3
ANALYSIS OF GROUND OPERATIONAL REQUIREMENTS AND PROBLEMS

The evaluation of ground operations and ground facilities requirements and problems is an important aspect of choosing a propellant for a new propulsion stage. In this study an analysis was performed to evaluate and compare operations, facilities and complexity for a mission of current interest and for several alternate propellants, and to compare propellant loading on the pad with loading in a remote explosive-safe facility. Ground rules and assumptions for the task included the following:

- Compare alternate propellant combinations including F_2/H_2, O_2/H_2, FLOX/CH_4, OF_2/B_2H_6, F_2/NH_3, and N_2O_4/A-50.
- Assume the propulsion stage is designed as a modular part of a Mars Orbiter spacecraft system to be launched by Titan IIID/Centaur from Complex 40-41 at Cape Kennedy.

Figure 35 shows a plan view of the Integrate Transfer Launch area at the Eastern Test Range (Cape Kennedy). These facilities were designed and built with the operational objective of accomplishing assembly and checkout of the complete vehicle, including spacecraft, in a central area. This central area, identified as the Vertical Integration Building (VIB), is shown in the figure. The functions performed in the VIB are:

- Assembly and checkout of the Titan core vehicle
- Assembly and checkout of the Centaur vehicle
- Assembly and checkout of the spacecraft and payloads vehicle
- Integrated mission simulation test

Upon completion of integrated mission simulated tests the Titan core/Centaur/ spacecraft vehicle is transported to the Solid Motor Assembly Building (SMAB) where the Titan solid motors are mated to the core vehicle. Finally, the complete, integrated vehicle is transported to the launch pad (40 or 41) for final pre-launch servicing, propellant loading, countdown, and launch.

ITL LAUNCH COMPLEXES 40 AND 41

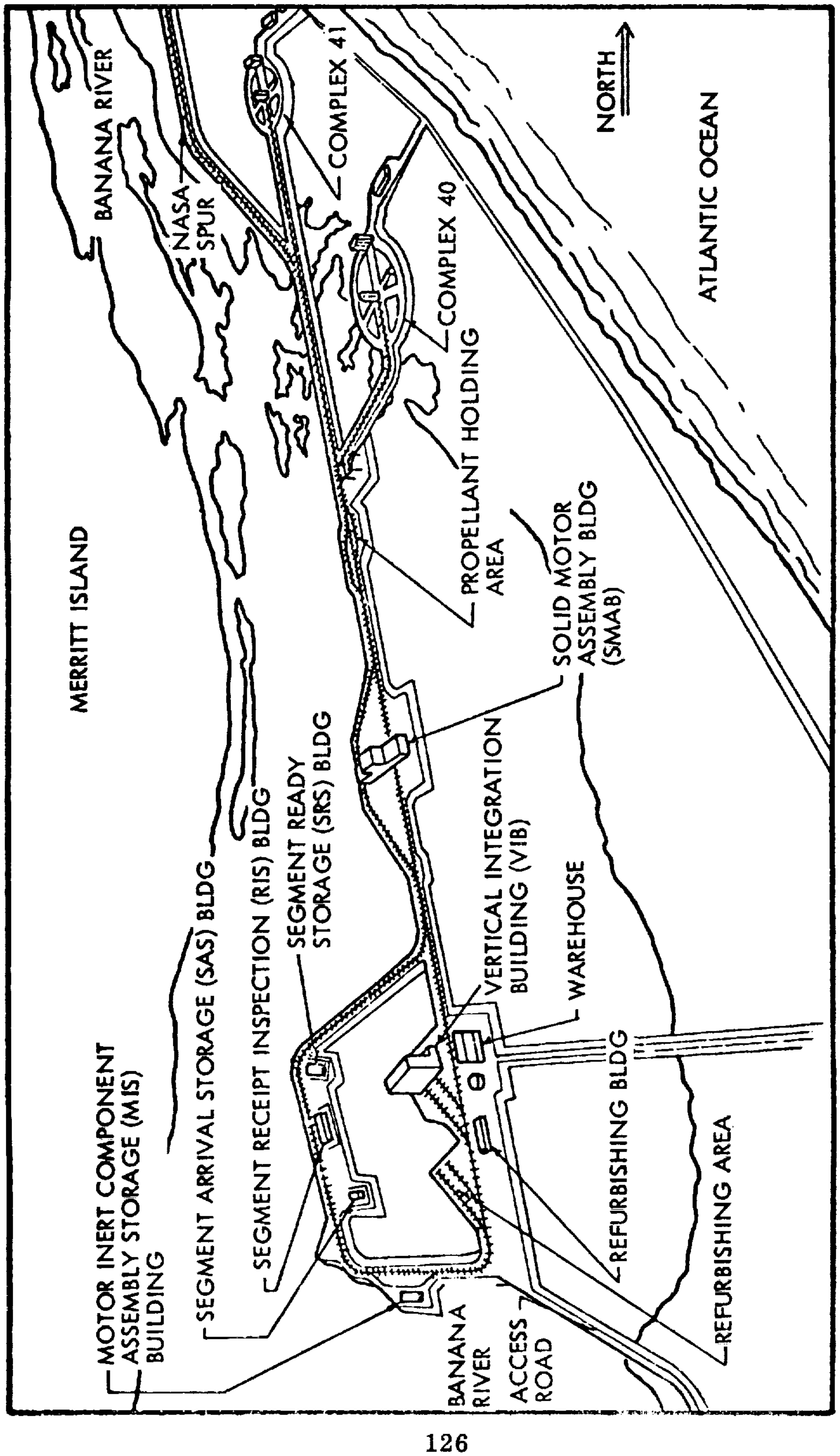


Fig. 35. Integrate Transfer Launch Area, Plan View

3.1 SUMMARY

No unusual problems were found in the handling of any of the candidates propellants at ETR. Complex 40, 41 at ETR can be readily modified for operations utilizing any propellant combinations studied. FLOX/CH_4 was found to be the most complex propellant from the standpoint of overall operations and facilities, and N_2O_4/A-50 the least complex. No-vent ground hold systems were found to be feasible for all propellants and to result in very small flight weight increments. Propellant loading of the spacecraft stage in a remote facility appears both feasible and desirable.

3.2 GROUND OPERATIONS

An analysis was made of the operational sequences required for propellant handling from initial tanking to final vehicle topping. This included studying problems associated with ground transport, storage at the launch complex, vehicle tank conditioning, transfer from storage to vehicle, thermal conditioning, insulation purging, and toxicity and hazard. Specific steps required for each of these items and each propellant are defined. An assessment was made of the requirements for a ground hold, and the recycle time and steps needed for a complete turnaround. Penalties associated with a no-vent condition on the pad were evaluated and a comparison made of propellant loading at the pad vs loading in a remote explosive-safe facility.

3.2.1 Launch Site Operations Flow

An analysis of the operational sequences related to propellant handling at the launch site was performed. Alternate sequences included propellant loading of the Mars Orbiter propulsion stage on the launch pad and loading in a remote facility prior to mating with the launch vehicle.

Figure 36 illustrates the launch site operations flow for the case where all propellant is loaded at the pad. Here propellant storage facilities for booster and spacecraft, tank conditioning facilities, and propellant loading facilities must be provided.

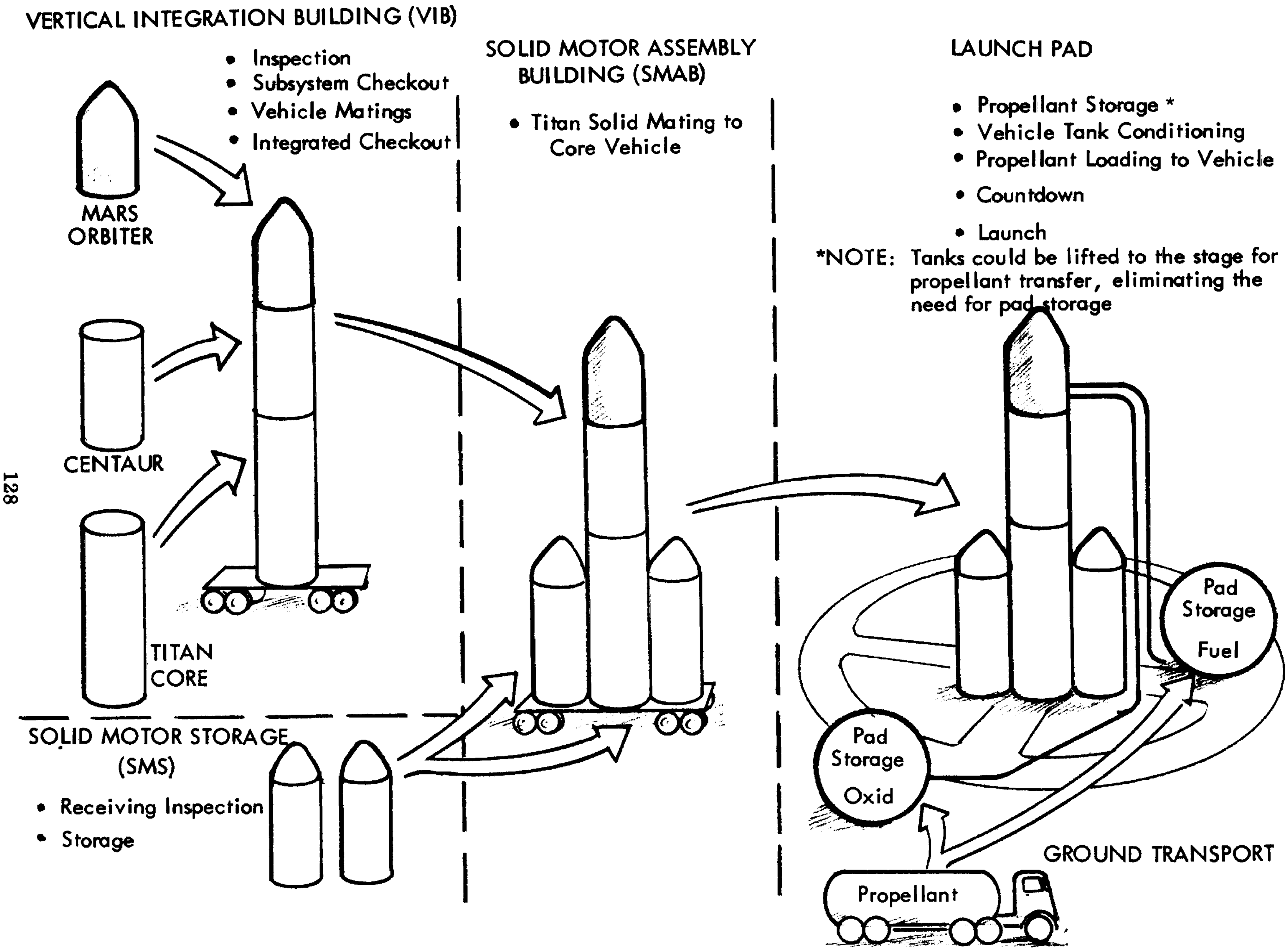


Fig. 36 Launch Site Operations – On-Pad Propellant Loading

Figures 37, 38, and 39 illustrate the launch site operations flow for concepts wherein the Mars Orbiter propellant tanks are loaded in a remote explosive-safe facility, checked out, and transported to the pad before mating with the launch vehicle. Figure 37 illustrates the sequence for an earth-storable propellant in the spacecraft with no special facilities required to maintain propellant temperatures during the period between load and launch. Figure 38 shows the addition of a liquid nitrogen cooling system that accompanies a spacecraft loaded with one of the space storable propellant combinations. Figure 39 shows the use of a closed loop helium heat exchanger for maintaining hydrogen in the liquid state, with liquid nitorgen used for conditioning the oxidizers. A comparison of on-pad and off-pad propellant loading is presented in the following section.

3.2.2 On-Pad Vs Off-Pad Propellant Loading

In an operational situation many advantages accrue to a spacecraft propulsion system that can be loaded, checked out, and buttoned up prior to moving the vehicle to the launch pad. This mode of operation has been used quite successfully for such spacecraft as Surveyor and Lunar Orbiter using earth-storable propellants. With the introduction of space-storable or cryogenic propellants in the spacecraft propulsion system, this approach is complicated by the need for thermal control to maintain the propellants in the liquid state. The advantages of off-pad propellant loading include:

- Launch pad occupancy time for spacecraft functional checks, leak checks, propellant tank preconditioning, etc., is considerably reduced.
- The possibility of delays or aborts on the launch pad being caused by propulsion system problems are minimized since final verification of readiness is with propellants loaded.
- Loading and checkout at a remote site is much more convenient and can be accomplished on a more leisurely schedule well before launch time.
- Backup spacecraft can be held in a completely loaded and checked-out state, ready to be transported to the pad in exchange for a malfunctioning vehicle.

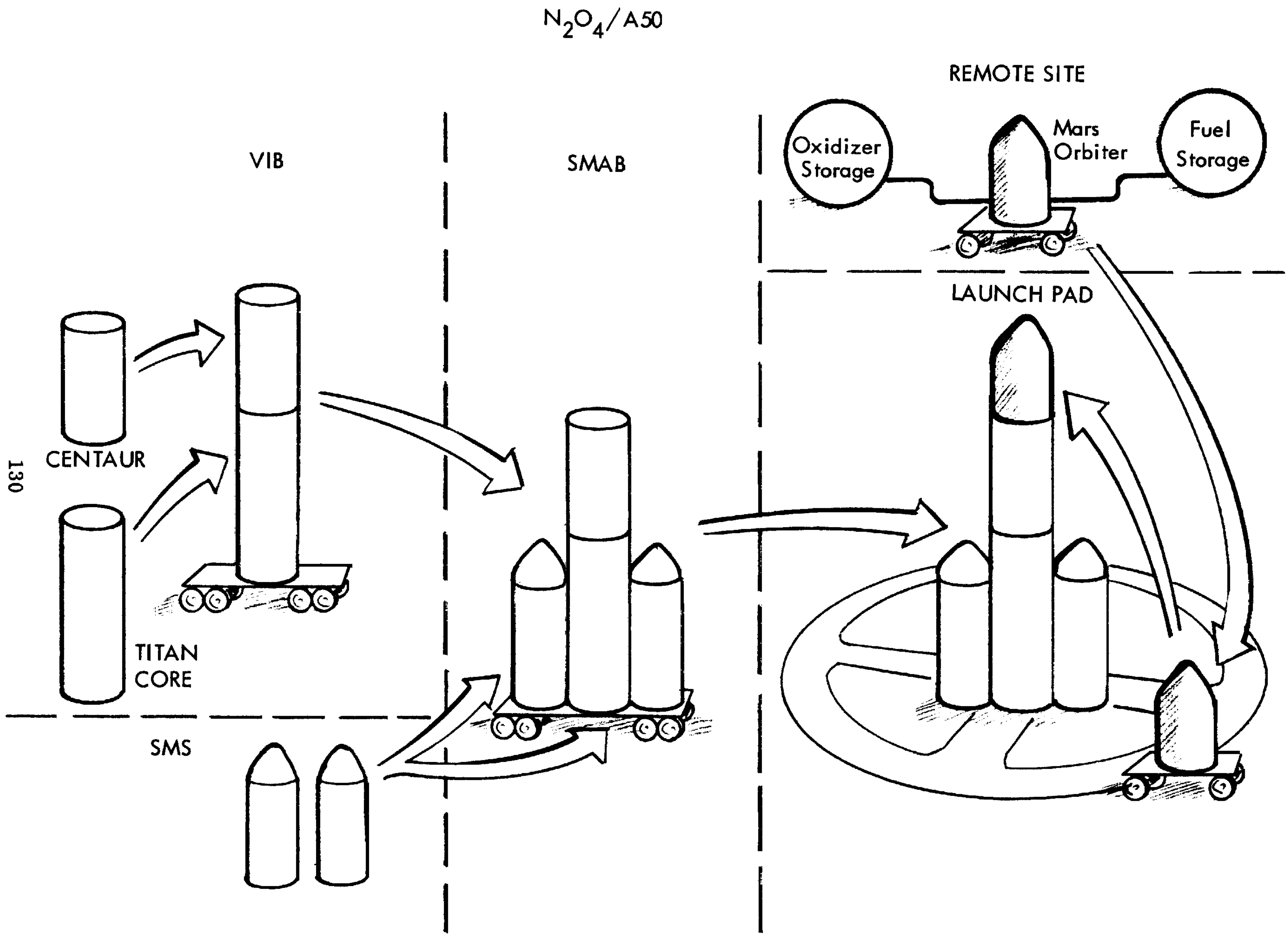


Fig. 37 Launch Site Remote Propellant Loading – N_2O_4/A-50

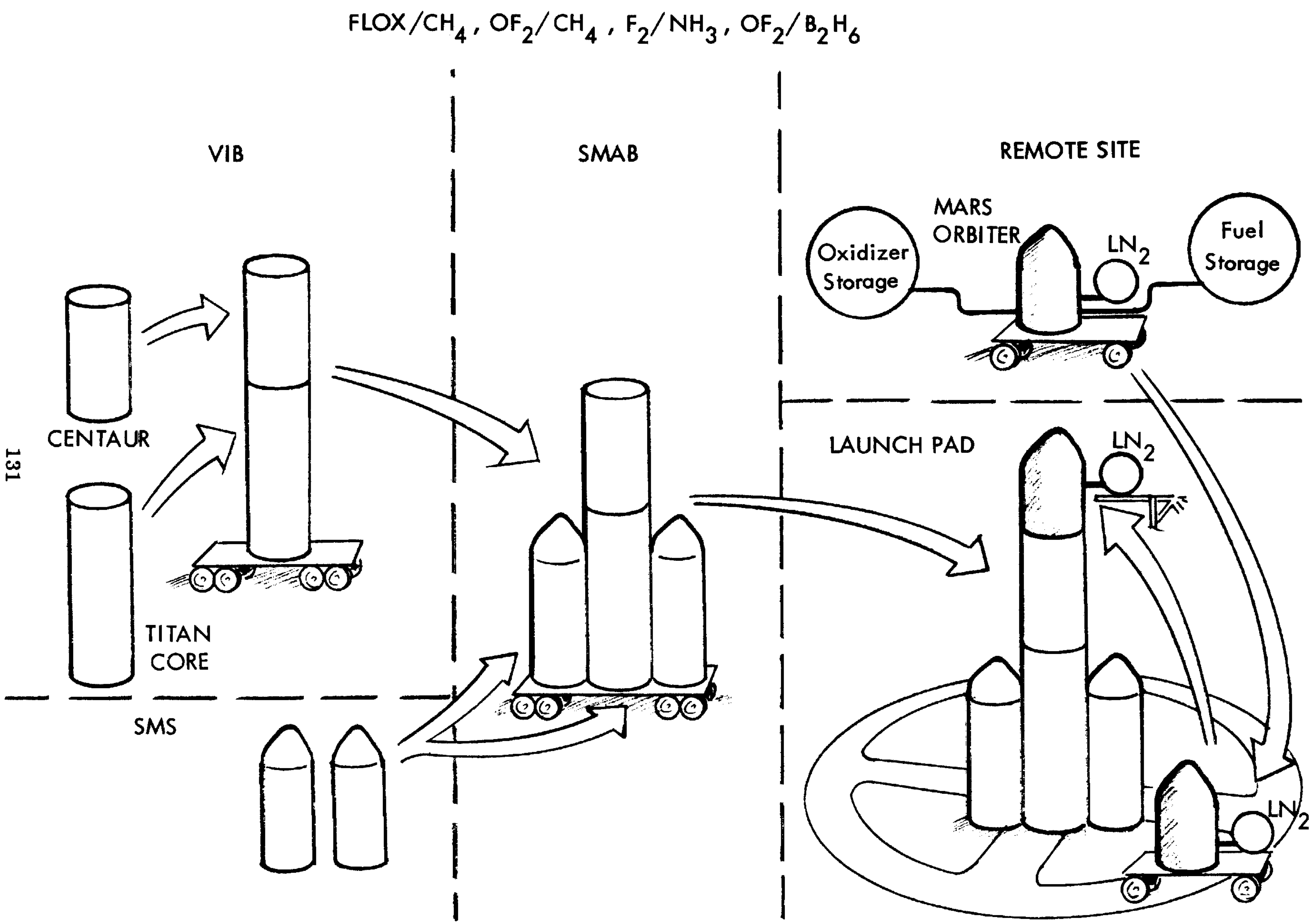


Fig. 38 Launch Site Remote Propellant Loading – FLOX/CH_4, OF_2/CH_4, OF_2/B_2H_6

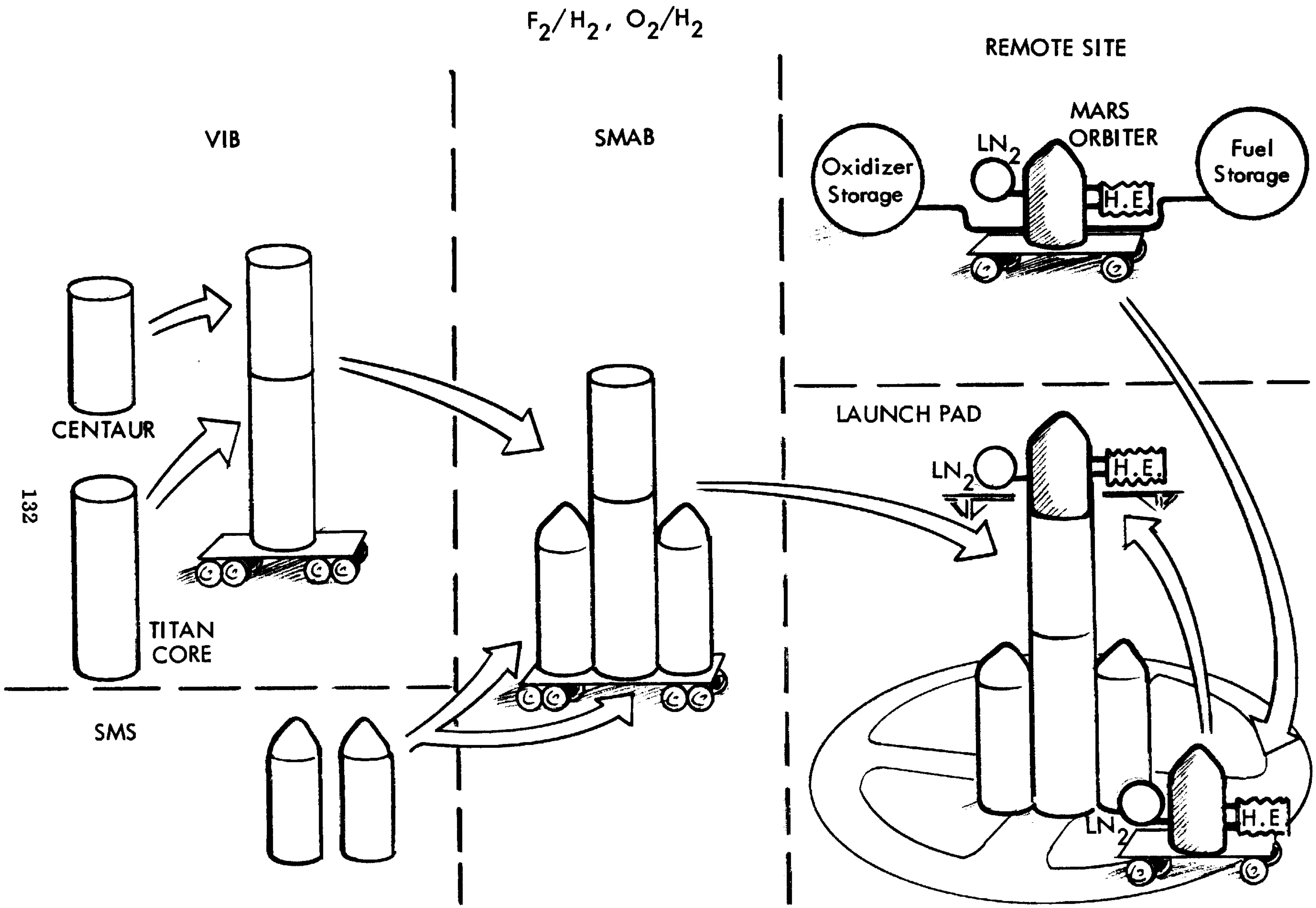


Fig. 39 Launch Site Remote Propellant Loading – F_2/H_2, O_2/H_2

- Provision of a thermal conditioning unit can eliminate the need for propellant venting, vent gas disposal, or propellant tank topping on the pad.
- The hazards of transferring spacecraft propellants at the pad are eliminated.

The disadvantages of off-pad propellant loading include:

- Thermal conditioning untis must be added for all space-storable and cryogenic propellants.
- Spacecraft flight weight will probably increase slightly to accommodate the flight portion of the thermal conditioning system.
- Handling of the fully loaded spacecraft from remote site to launch vehicle mating will be more difficult due to the increased weight and to the greater hazards introduced.

The final decision to load propellants on-pad or off-pad will require a more complete definition of the vehicle system. It appears that off-pad propellant loading for spacecraft of the Mars Orbiter size, using any class of propellant, is an attractive goal. The key problem lies with the design of the propellant thermal control system.

Concepts for non-vented ground hold thermal conditioning systems are illustrated in Figures 40, 41, and 42. Figure 40 shows a liquid N_2 spray cooling concept for ground control of the space-storable propellants. A similar concept has been studied in detail by TRW Systems under contract NAS 7-711 to NASA-JPL. This contract has been completed and a final report, TRW 111455-6013-R0-00, has been issued.

Figure 41 shows a cold He heat exchanger concept for thermal control of liquid H_2. Here the H_2 is recirculated through a closed loop refrigerator which is disconnected just prior to liftoff. Additional hold time between refrigerator disconnect and booster liftoff can be gained with the concept shown in Figure 42. Here H_2 is subcooled or slushed and maintained in this condition by recirculation through an external subcooled or slush H_2 generator. This concept could also be used to provide subcooled or slush H_2 for the reduced weight or extended mission time benefits to be gained on space missions.

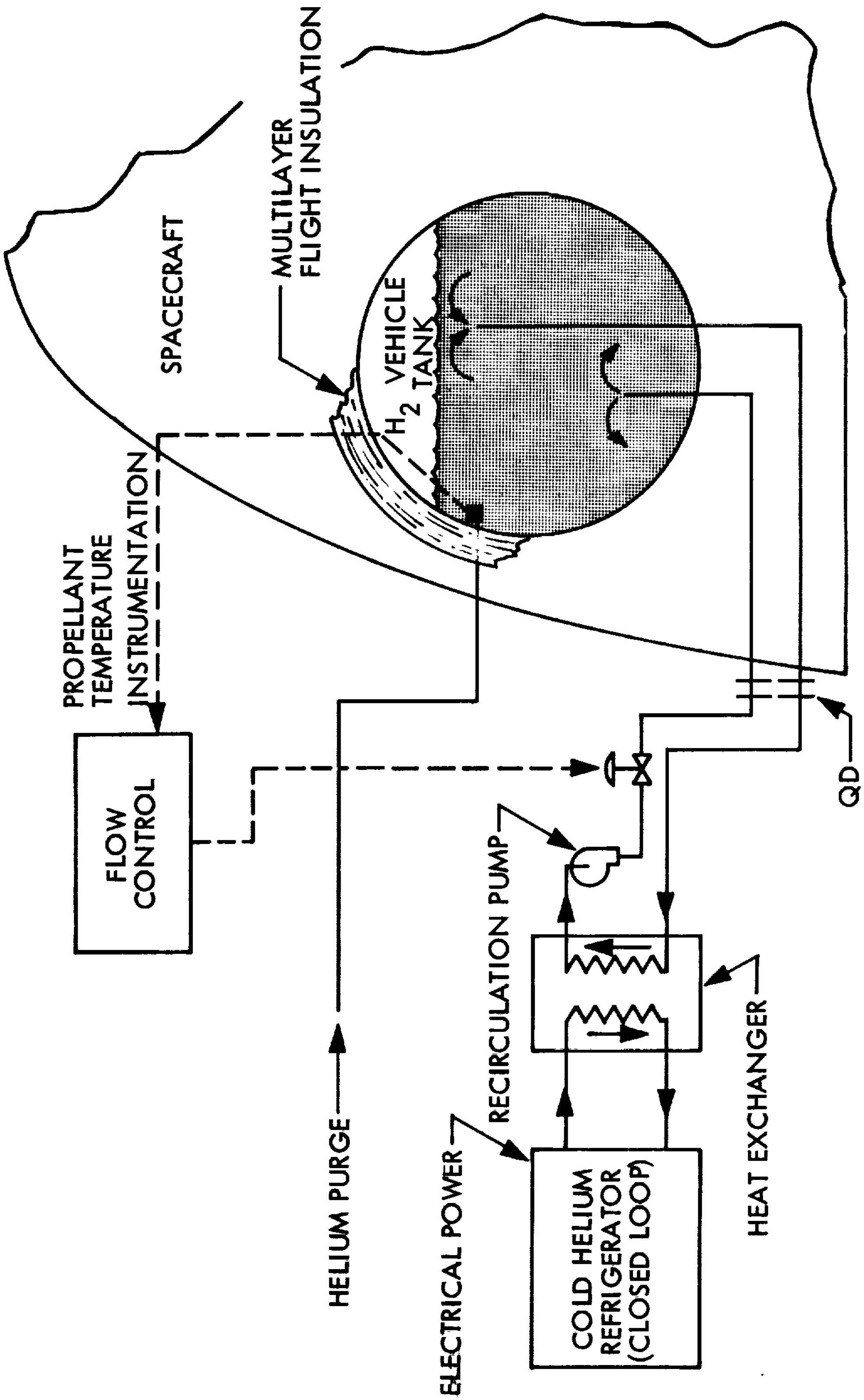


Fig. 41 Ground Hold Non-Vented System – Cold He Heat Exchanger Concept for LH_2

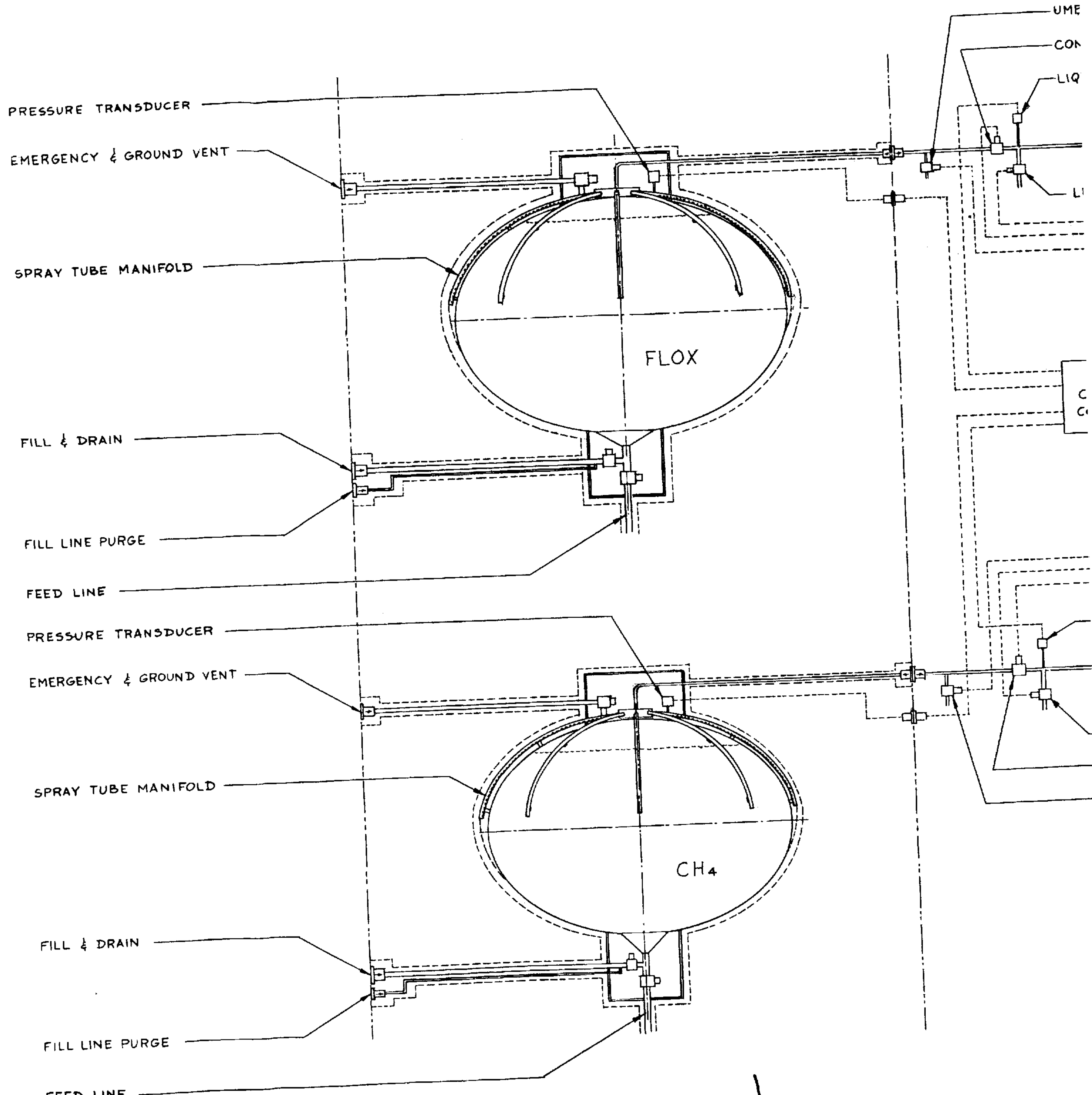


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Fig. 40 G
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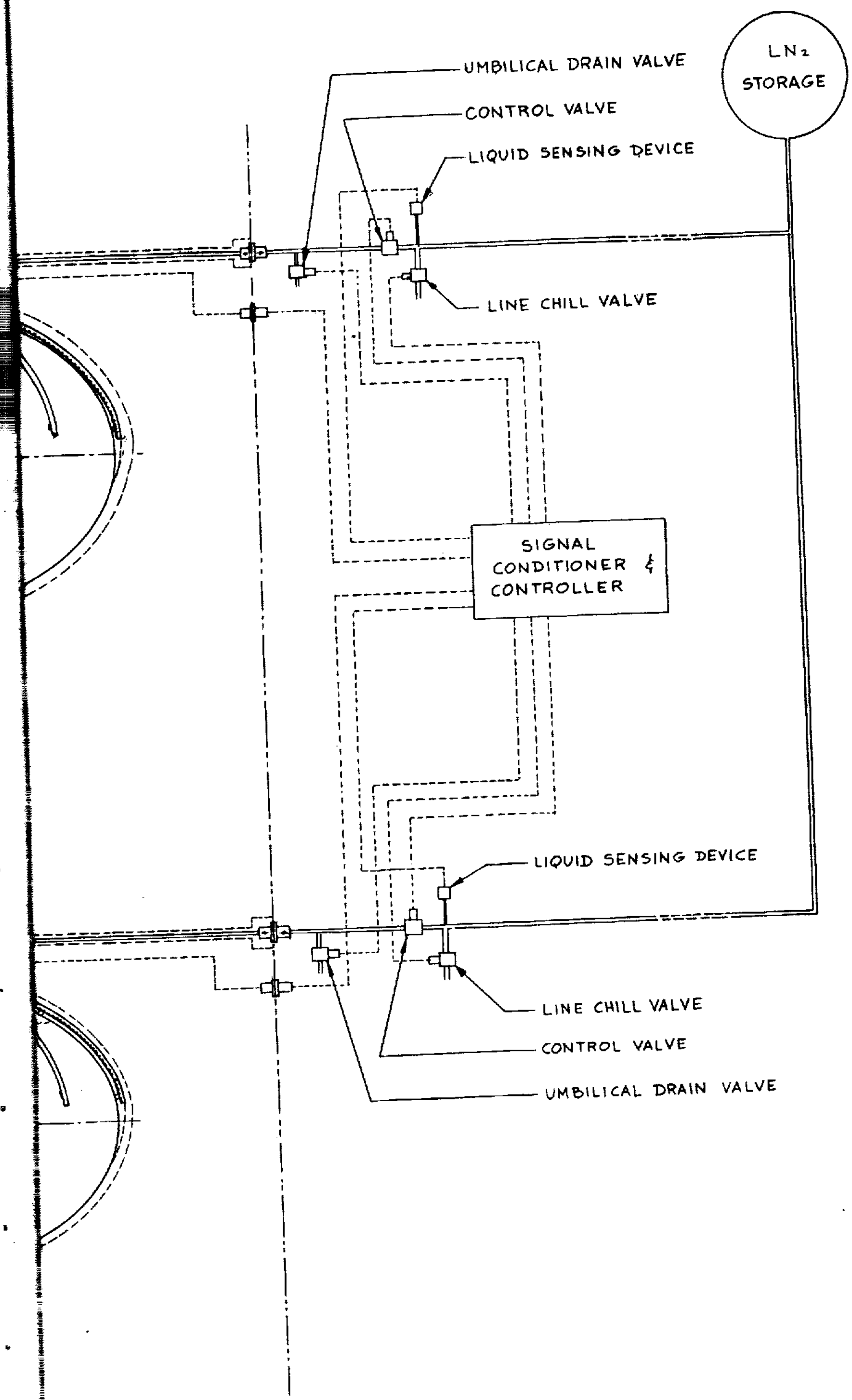

Fig. 40 Ground Hold Non-Vented System LN_2 Spray Cooling Concept for NH_3, B_2H_6, CH_4, OF_2, O_2, FLOX, F_2

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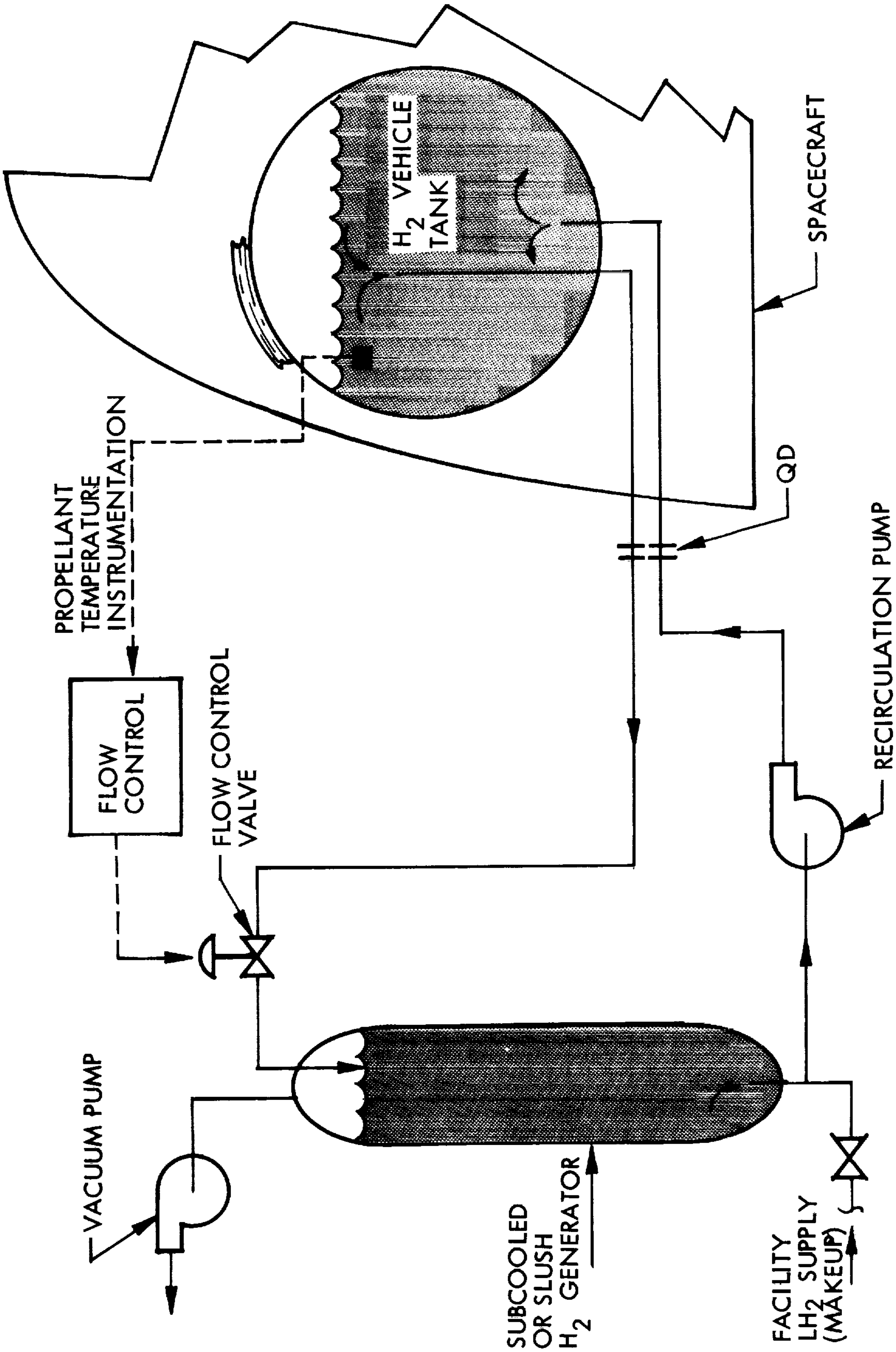


Fig. 42 Ground Hold Non-Vented System – Subcooled/Slush Concept for LH_2

3.2.3 Analysis of Ground Hold Heating and No-Vent

Ground hold heating analyses were conducted in order to determine heat input to the propellants, boiloff rates for vented non-cooled propellants, and cooling requirements and weight penalties for non-vented propellants. It was assumed that a non-vented system accompanied off-pad propellant loading, and that systems loaded on-pad were vented.

Heating calculations were based on tank sizes, insulation thicknesses, and configurations developed during Phase I of the study and documented in Lockheed report K-19-68-6, Volume II, NASA CR 96988. Configuration data are presented in Table 24. Nitrogen gas was assumed used to purge the shroud and propellant tank insulation except for hydrogen tanks. Hydrogen tank insulation was assumed purged with helium to avoid freezing the purge gas.

Results of the ground hold heating analysis are presented in Table 25 including heat inputs, boiloff rates for vented systems, and cooling requirements for non-vented systems. Heat inputs include convective heat transfer in the shround and insulation thermal conductivities corresponding to the type of purge gas used. Tank areas used in the analysis were considered to be mean heat transfer areas depending on insulation thicknesses as well as tank size. Heat inputs range from zero for the earth storable to a high of 25,590 Btu/hr for hydrogen.

Ground hold propellant vent rates were calculated by assuming that all heat input was translated into propellant boiloff, with saturated propellant at one atmosphere pressure. Boiloff rates range from zero for N_2O_4/A-50 to 132 lb/hr for H_2, with O_2 and the fluorinated oxidizers all venting at the rate of tens of pounds per hour. Vent gas disposal facilities, plus tank topping facilities, will be required for all of the space storables and the cryogens.

In analyzing the non-vented systems, active cooling was provided to remove all incoming heat. A relatively simple nitrogen cooling system was assumed used on

Table 24

CONFIGURATION DATA FOR GROUND HOLD ANALYSIS

Propellant	Number of Tanks	Size of Tanks and Shape	*Insulation Thickness Inches	Shroud Purge	Insulation Purge
F_2	2	46.7" D Sphere	1/2	GN_2	None
H_2	1	82.6"/59" ellipsoidal	1-3/4	GN_2	GHe
O_2	4	40.6" D Sphere	1/2	GN_2	None
H_2	1	104" × 74.6" ellipsoidal	1-3/4	GN_2	GHe
FLOX	2	47.8" D Sphere	1/2	GN_2	None
CH_4	2	40.8" D Sphere	1/2	GN_2	None
OF_2	2	46.8" D Sphere	1/2	GN_2	None
CH_4	2	41.5" D Sphere	1/2	GN_2	None
F_2	2	46.2" D Sphere	1/2	GN_2	None
NH_3	2	40.3" D Sphere	1/2	GN_2	None
OF_2	2	48.2" D Sphere	1/2	GN_2	None
B_2H_6	2	44.7" D Sphere	1/2	GN_2	None
N_2O_4	2	48.2" D Sphere	1/2	None	None
A-50	2	44.7" D Sphere	1/2	None	None

*Optimized insulation table 31, Vol. II K-19-68-6 and table 14, Vol. III K-19-68-6 (worst case for ground hold, due to minimum insulation thickness)

Table 25

GROUND HOLD ANALYSIS RESULTS

		VENTED SYSTEM	NON-VENTED SYSTEM		
PROPELLANT	GROUND ENVIRONMENT HEAT INPUT (BTU/HR)	BOILOFF RATE, (LB/HR)	COOLING METHOD	COOLING FLOWRATE (LB/HR)	ESTIMATED FLIGHT HARDWARE INCREASE (LB)
F_2	6,100	82.4	LIQUID N_2 SPRAY COOLING	72.5	2.0
H_2	16,300	84.0	*COLD HELIUM	685.0	6.0
O_2	9,220	100.4	LIQUID N_2 SPRAY COOLING	109.5	3.1
H_2	25,590	131.8	*COLD HELIUM	1070.0	6.0
FLOX	6,364	85.4	LIQUID N_2 SPRAY COOLING	75.6	2.0
CH_4	4,302	17.4		51.2	1.6
OF_2	5,232	58.6		62.1	2.0
CH_4	4,460	18.0		53.0	1.6
F_2	5,984	80.6		71.1	1.9
NH_3	1,212	2.1		14.4	1.5
OF_2	5,538	62.0		65.8	2.0
B_2H_6	3,456	15.6		41.1	1.7
N_2O_4	NEGLIGIBLE	NEGLIGIBLE	NONE	NONE	NONE
A-50	NEGLIGIBLE	NEGLIGIBLE	NONE	NONE	NONE

*CLOSED CYCLE HELIUM HEAT EXCHANGER WITH NO REFRIGERANT LOSS all tanks except for hydrogen. This system was illustrated earlier in Figure 40. The liquid nitrogen is distributed uniformly over the tank surface under the insulation through a system of small tubes equally spaced around the tank. The liquid nitrogen evaporates upon contacting the warmer tank wall and insulation, absorbing heat during the transition from liquid to gas. The liquid N_2 cooling flowrates are shown on Table 25 and range between 14 and 110 pounds per hour for the space storables.

The hydrogen tanks are maintained in a no-vent condition at atmospheric pressure by circulating the liquid hydrogen through a helium cooled heat exchanger, illustrated earlier in Figure 41. Here the helium flow rate requirements are high, but since this is a closed loop system the helium is not lost. An estimate of the helium refrigeration power requirement was made based upon the H_2 heat loads applicable to the F_2/H_2 and O_2/H_2 systems. With a coefficient of performance of about 0.03, which is typical of helium refrigeration systems, the power requirement for sustaining the H_2 in the F_2/H_2 system is about 165 kw; and for the O_2/H_2 system about 250 kw. This implies the need for a relatively large unit considering that it must be mobile to accommodate transfer to the pad. Characteristically, maintenance and reliability of helium referigerators would dictate the need for a reducndant unit for emergency standby.

An alternate approach to use of the helium refrigerator is use of an expendable H_2 heat exchanger In this system the ground based heat exchanger contains LH_2 A vacuum pump maintains a low pressure on the ground side of the heat exchanger so that the LH_2 temperature is about 26°R. LH_2 from the spacecraft (37°R) is then circulated through the ground heat exchanger and cooled. This system would require a ground system pumping power level of about 15 to 20 kw and would expend about 90 pounds of H_2 per hour for the F_2/H_2 system and about 140 pounds of H_2 per hour for the O_2/H_2 system.

The estimated increases in flight hardware weight to provide a non-vented system are presented in Table 25. These weights are quite modest with a maximum of 6 pounds for hydrogen, assuming that the flight-designed tank insulation does not require modifications.

3.2.4 FLOX Differential Boiloff Analysis

In a vented FLOX system the fluorine and oxygen in the mixture will boil off at different rates, thus changing the mixture ratio. An analysis was performed to determine the rate of FLOX boiloff and the change in mixture with time during vented ground hold operation.

The differential boiloff problem results from the difference in the composition of the boiloff vapor as compared to the liquid. Fig. 43 shows the relationships between percent fluorine in the mixture and the boiling point temperature for liquid and for vapor, and indicates that the fluorine will boil off at a higher rate than the oxygen. Figure 44 presents the heat of vaporization as a function of composition and boiling temperature. Using the information in Figures 43 and 44 the history of the composition of the liquid FLOX was determined as a function of time under ground hold conditions for the Mars Orbiter.

The calculations were based on the Phase I study configuration with FLOX (82.6% F_2)/ CH_4 propellants. The initial propellant load was 5525 lb and the estimated ground hold heat rate 6346 Btu/hr. Figure 45 shows the percent of LF_2 in the FLOX and the total mass remaining in the tanks as a function of ground hold time. During the first six hours of heating the change in composition is less than 1%. As heating continues the change becomes progressively larger. It appears that the changes could be tolerated for short hold periods, but that correction of the mixture during topping, or complete replacement of the propellant, would be required if the hold time is long.

3.2.5 Propellant/Propulsion Systems Operations Comparisons

An operations comparison was made for the on-pad propellant loading concept in order to evaluate the effects of propellant choice on propulsion system related activities and operations time requirements. The results of this comparison are summarized in Figure 46. The following bases and assumptions were made in preparing the chart:

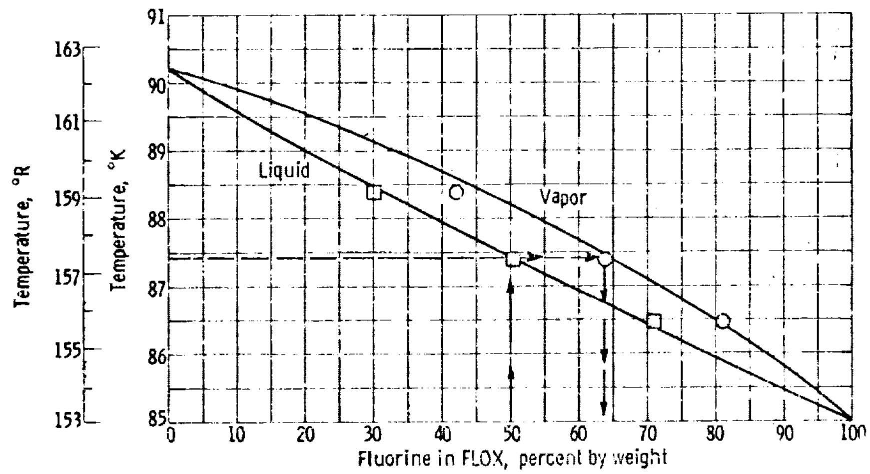


Fig. 43 Boiling Point Composition Diagram for FLOX Mixtures

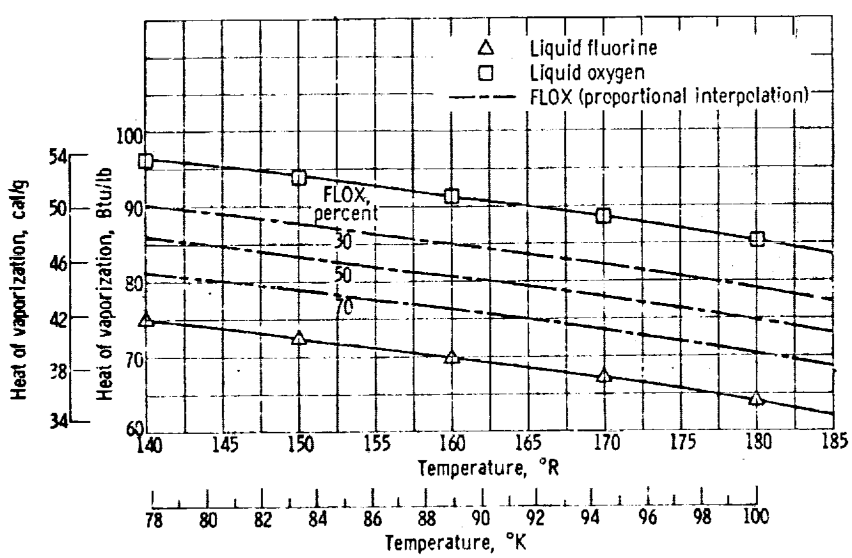


Fig. 44 Heat of Vaporization of Liquid-Fluorine-Liquid-Oxygen Mixtures

NOTE: Figs. 43 and 44 reproduced from: Handling and Use of Fluorine and Fluorine-Oxygen Mixtures in Rocket Systems, Harold W. Schmidt, Lewis Research Center, 1967, NASA Sp-3037.

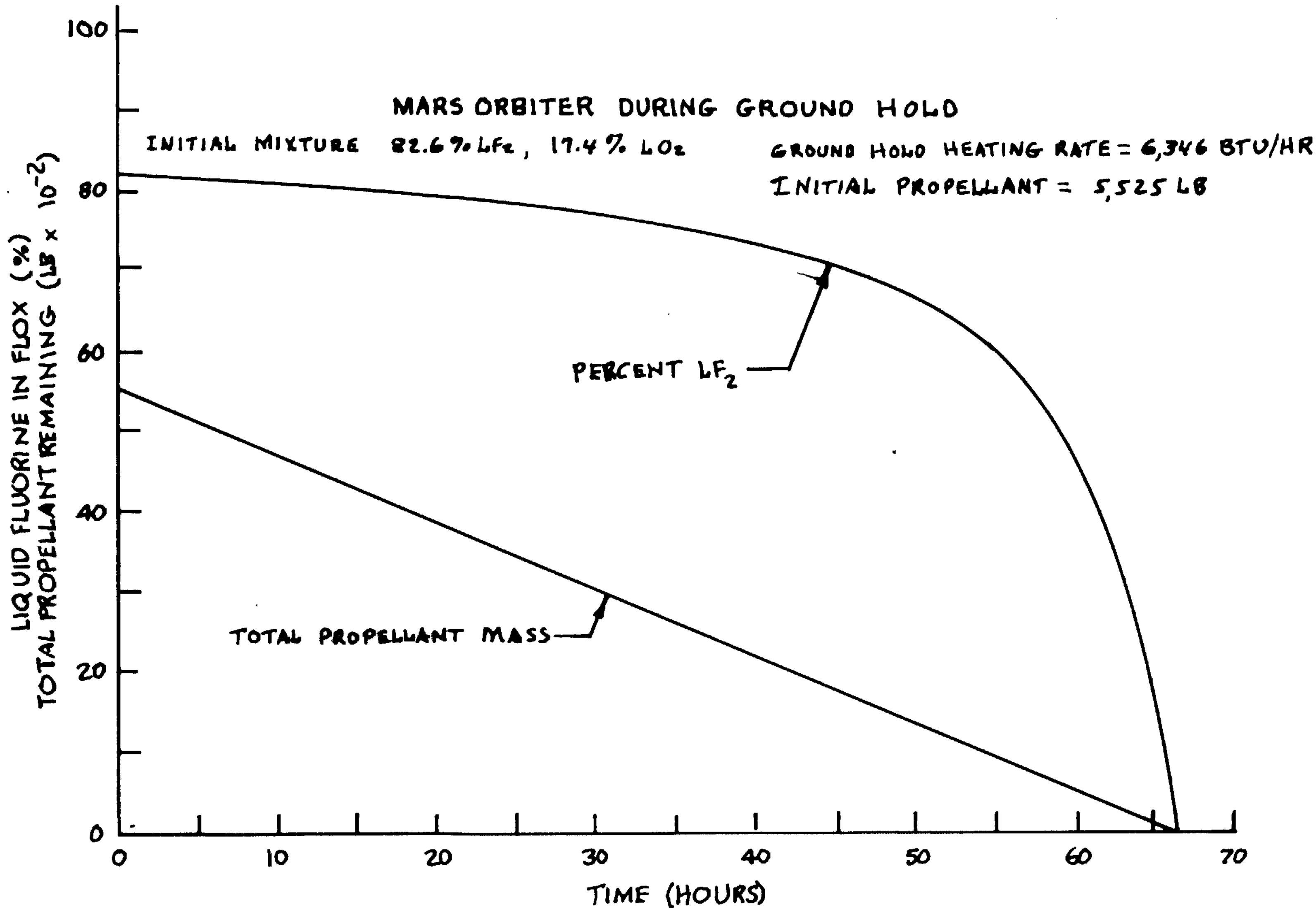
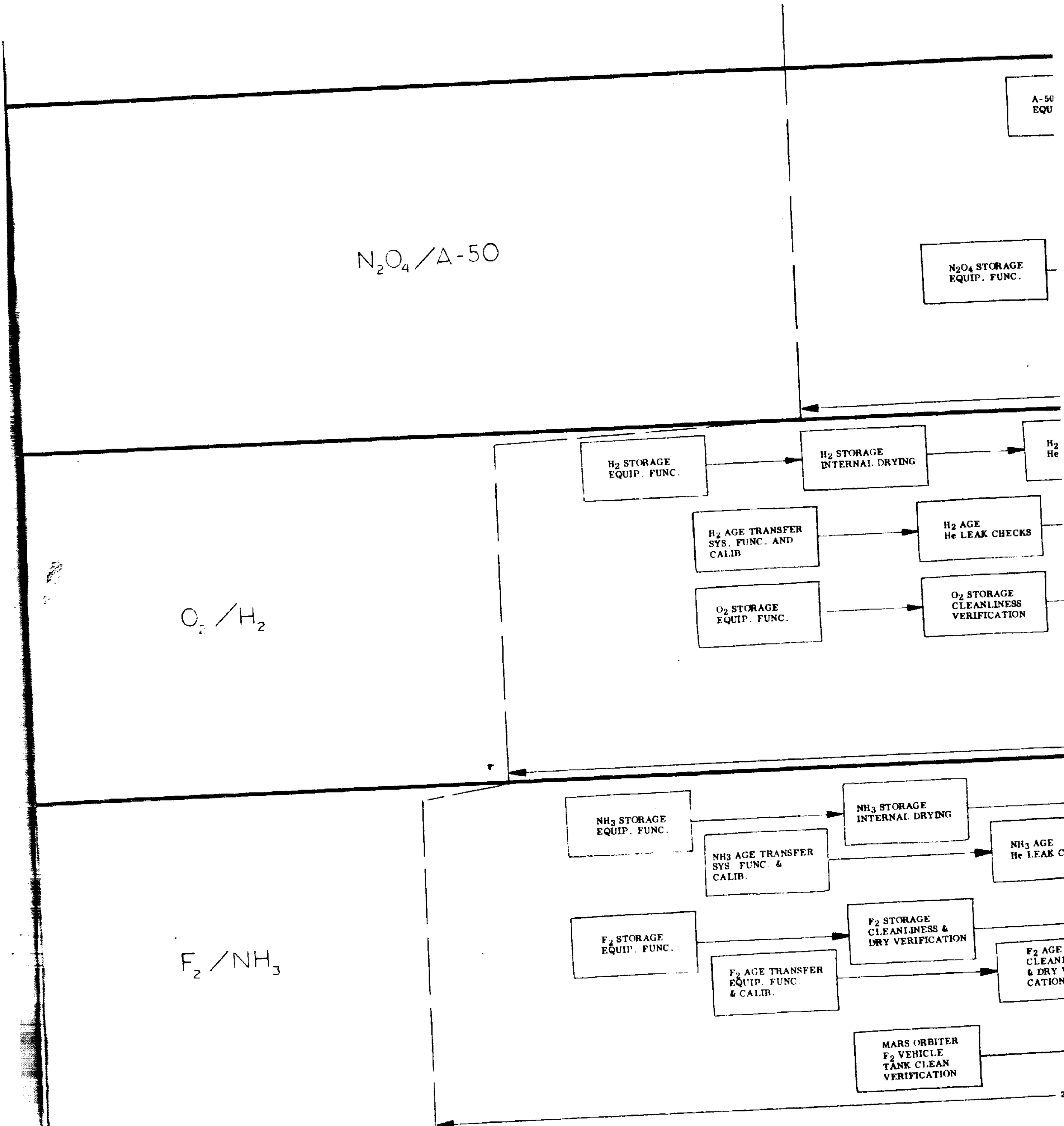


Fig. 45 FLOX Differential Boiloff Effects



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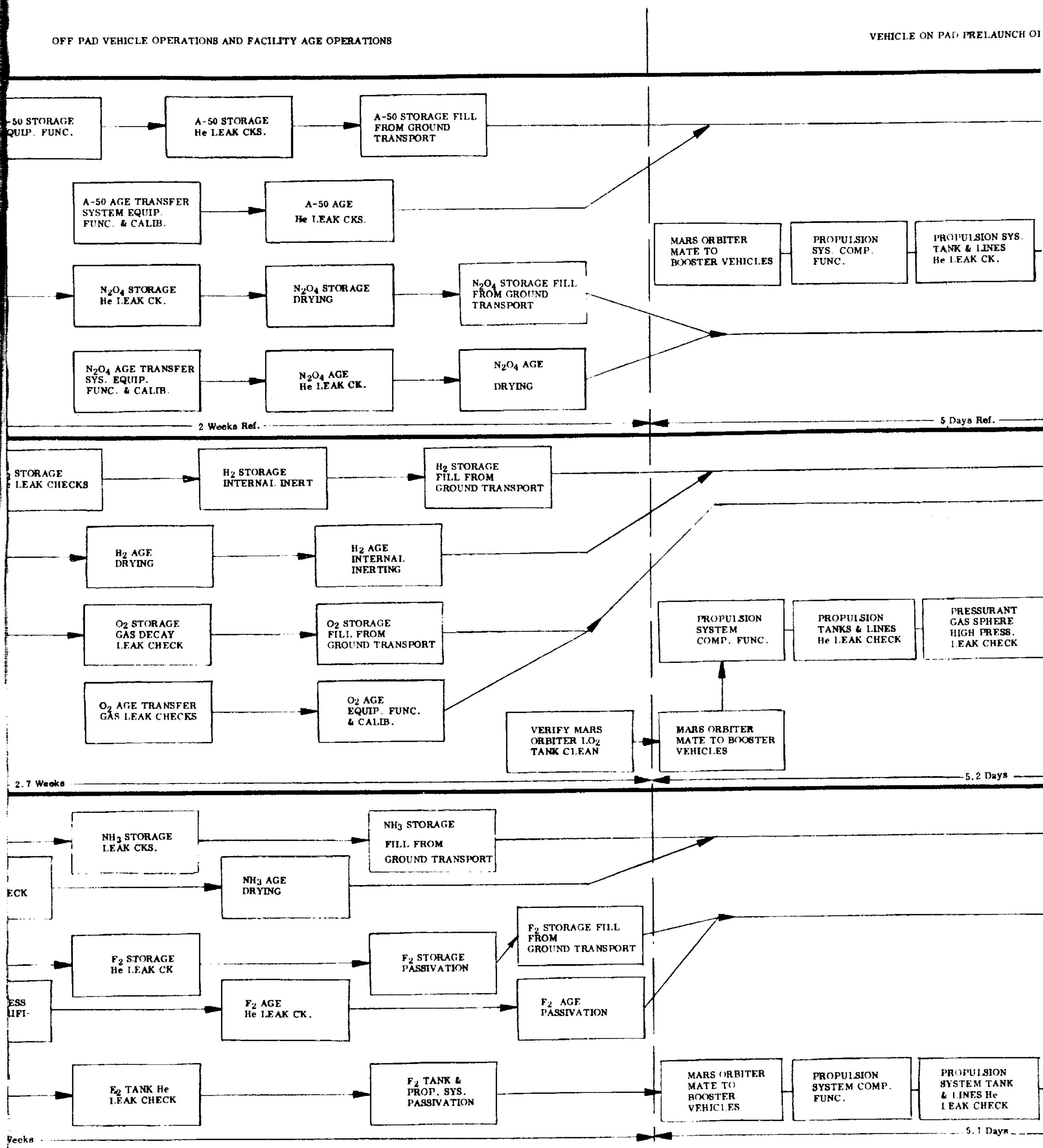

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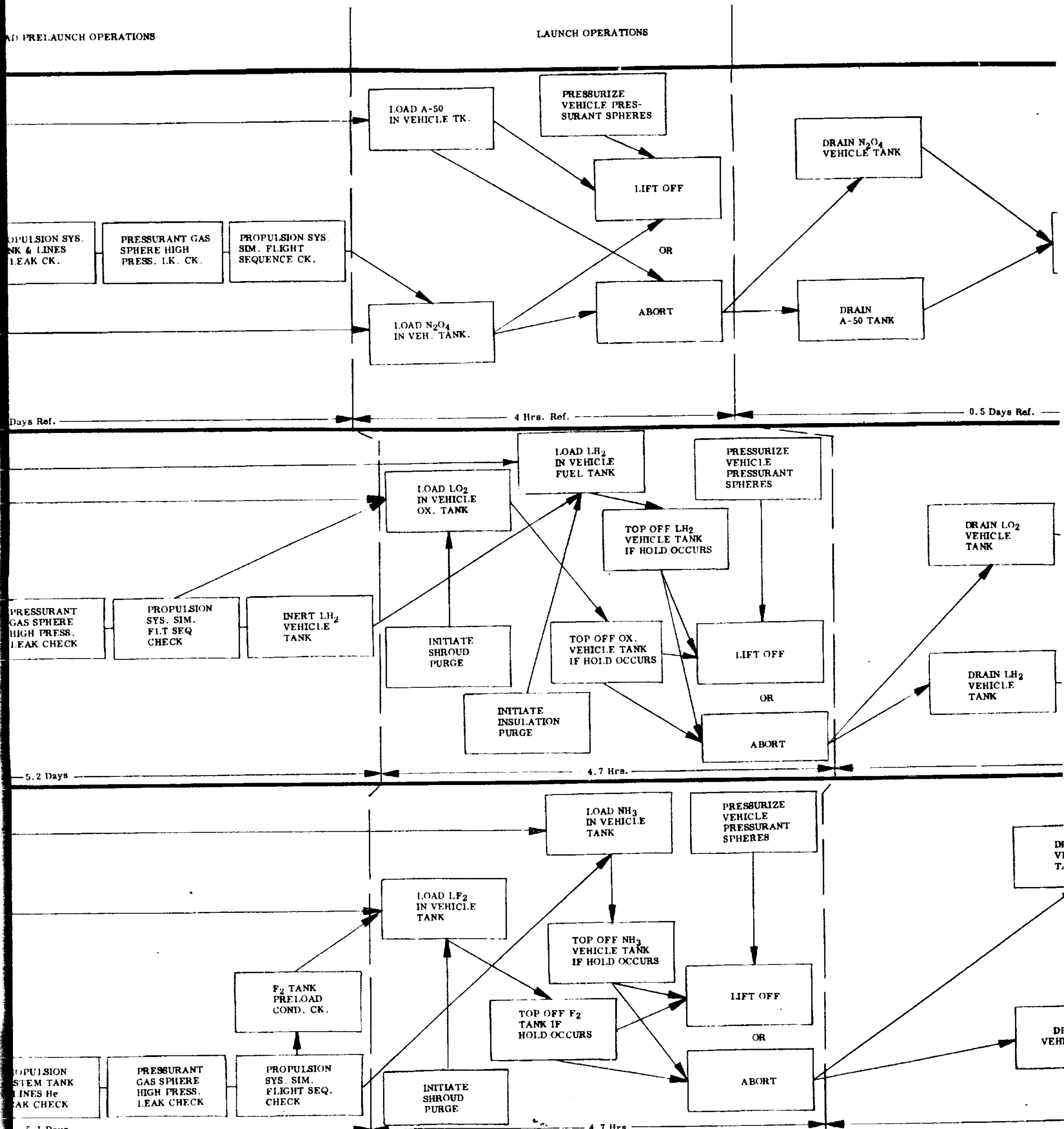

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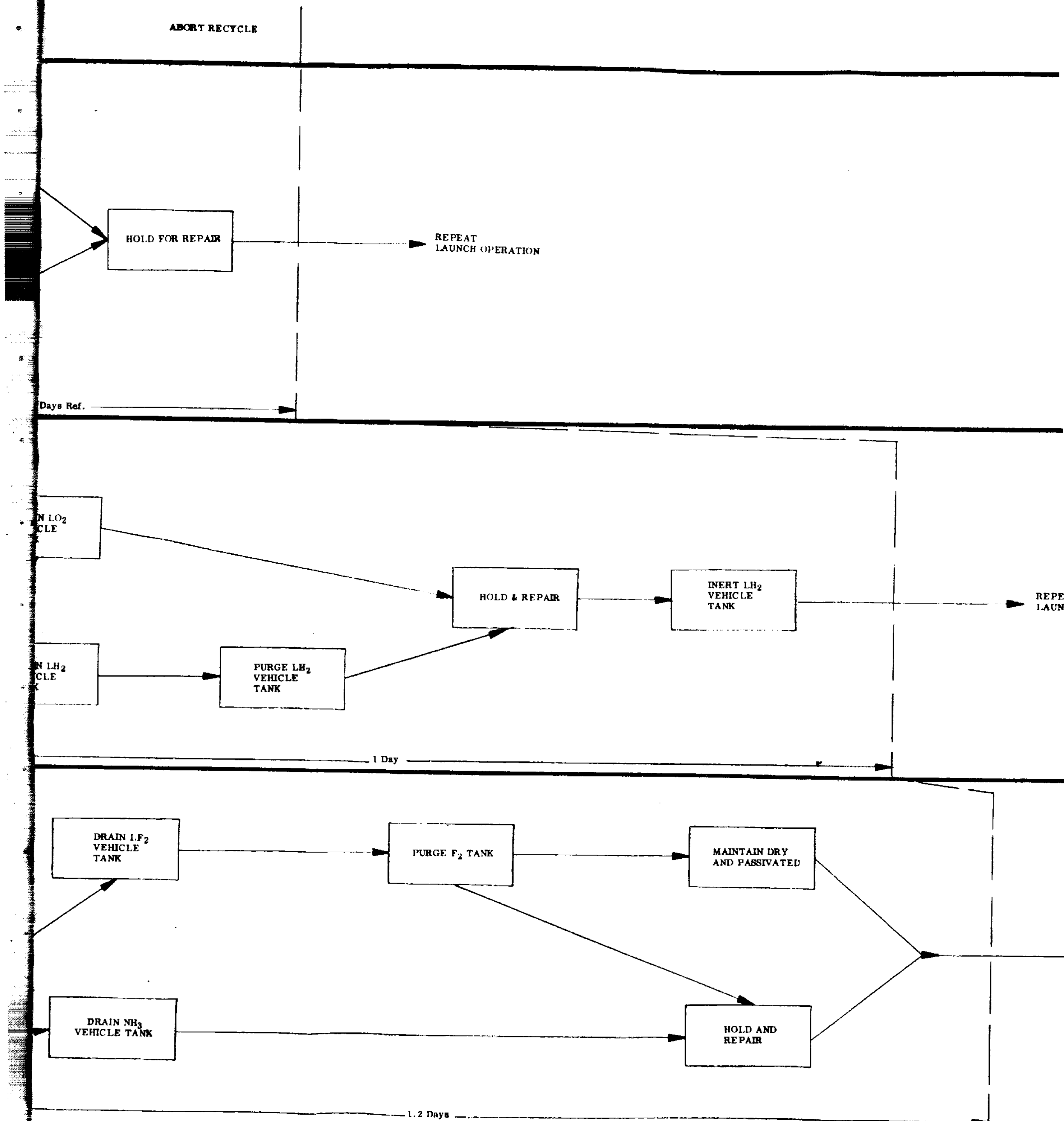

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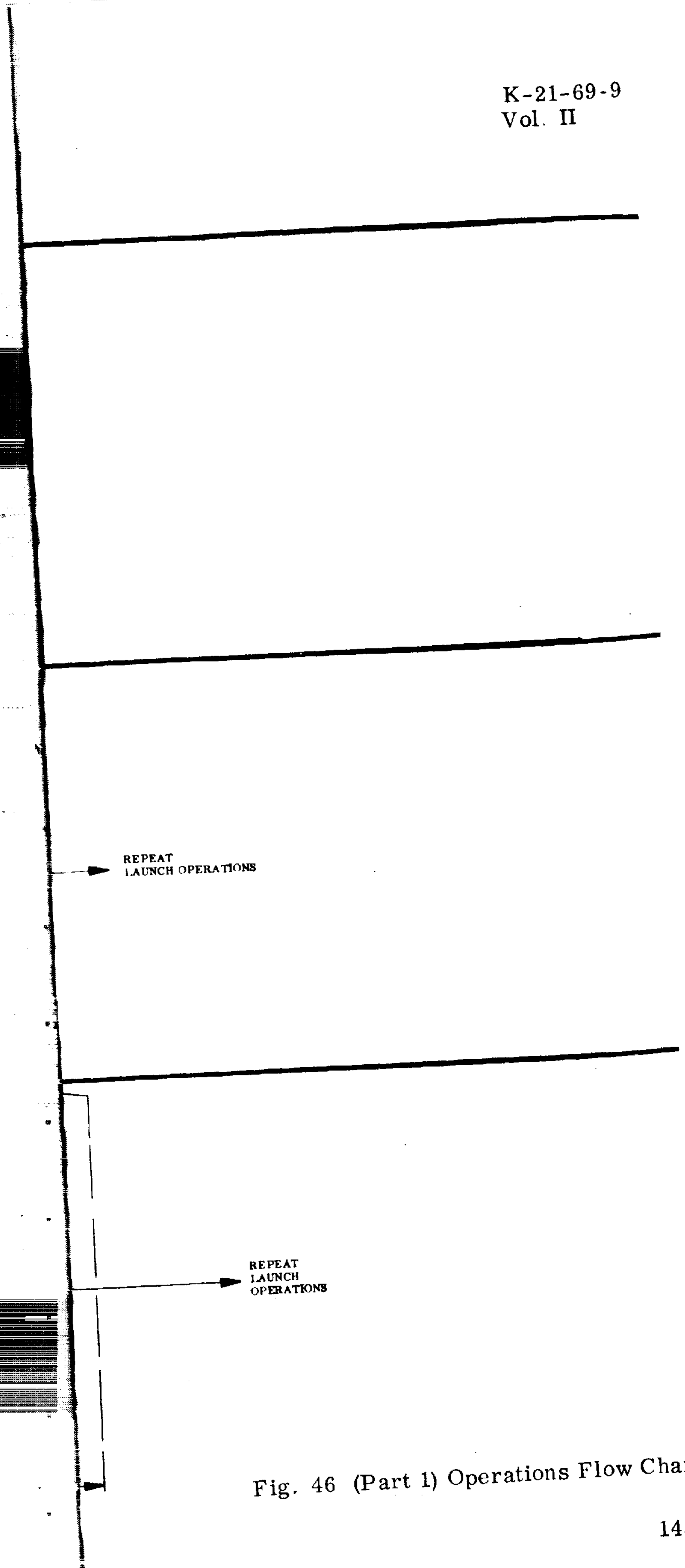
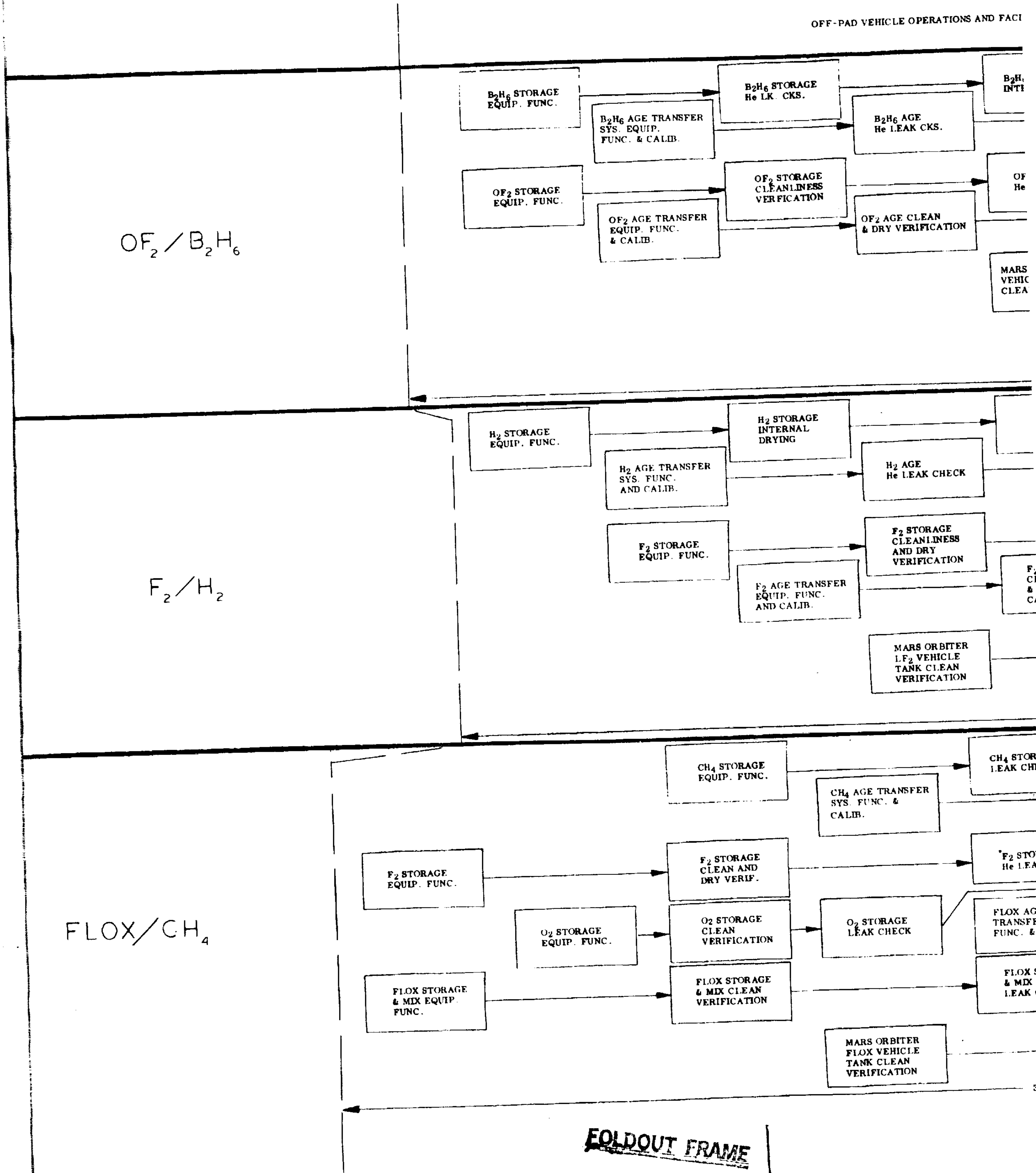


Fig. 46 (Part 1) Operations Flow Chart



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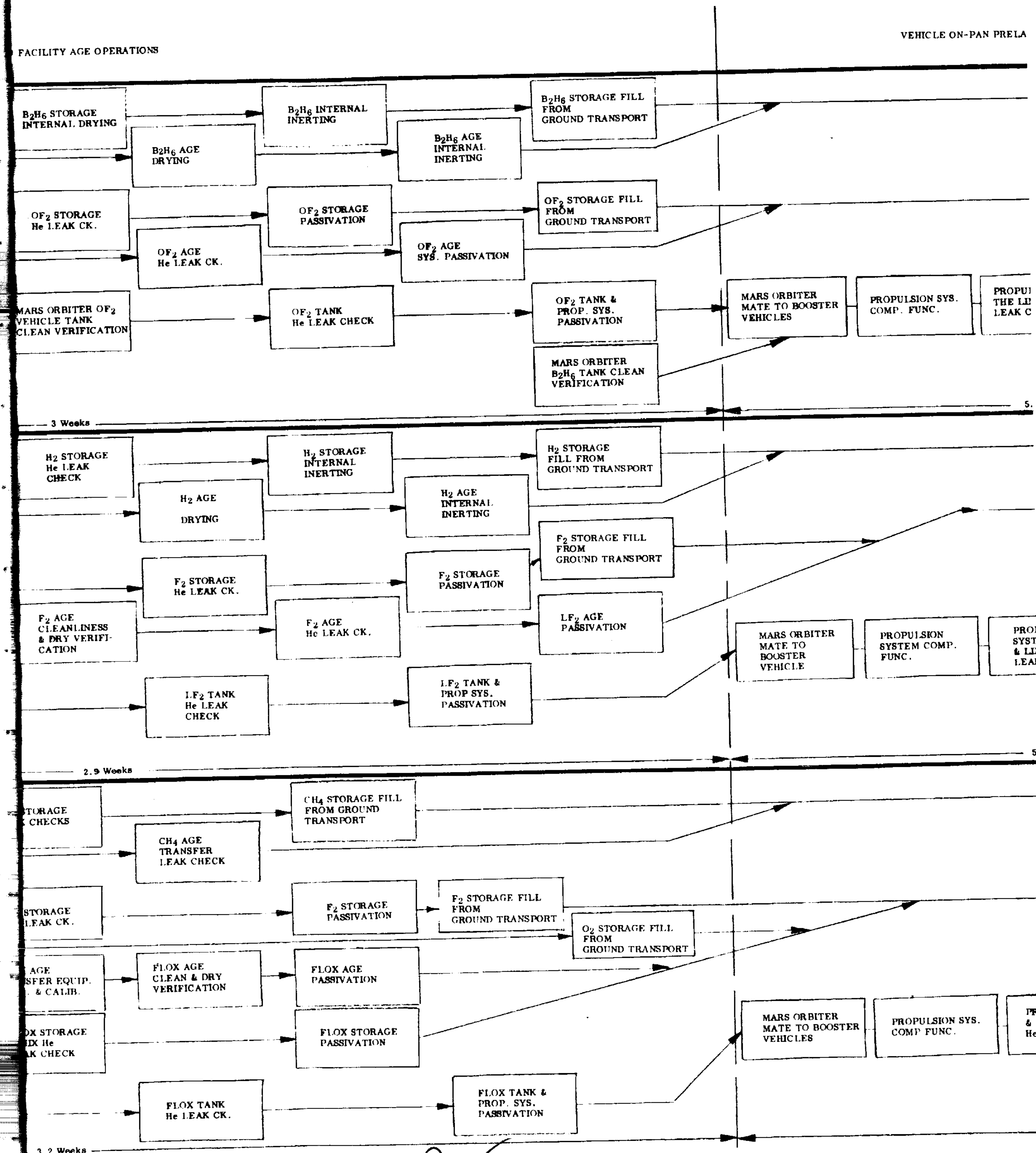

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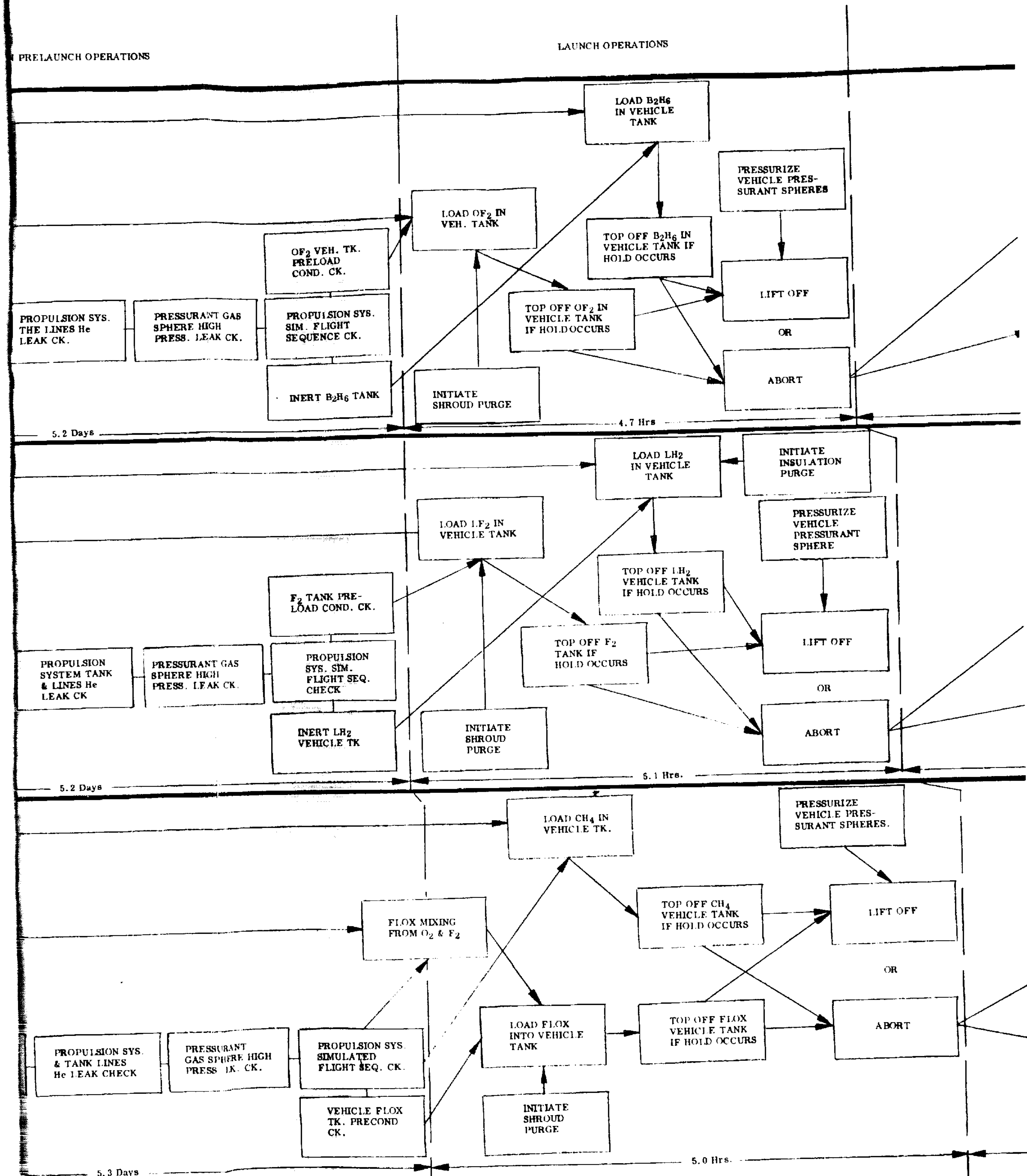

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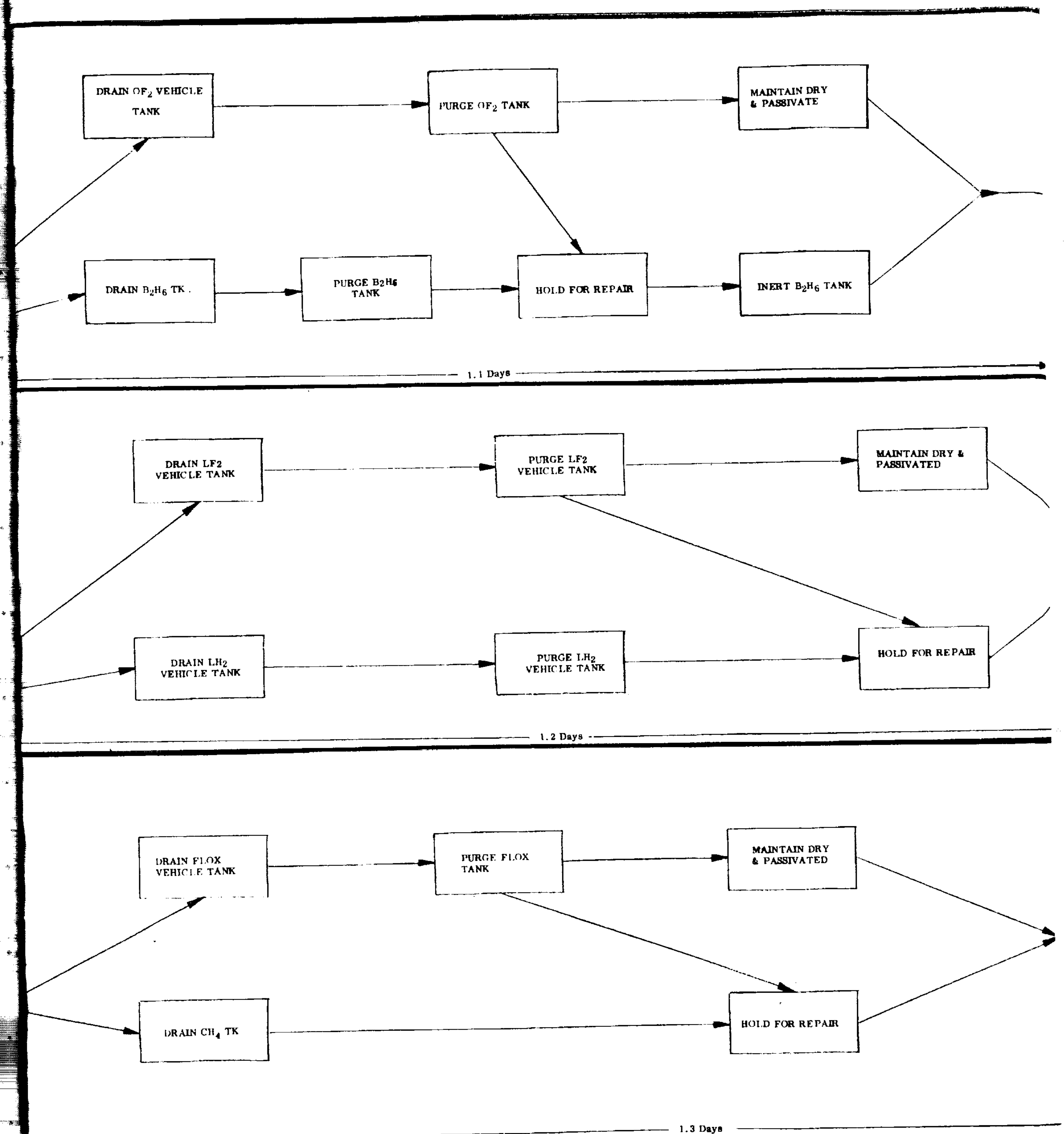
ABORT RECYCLE



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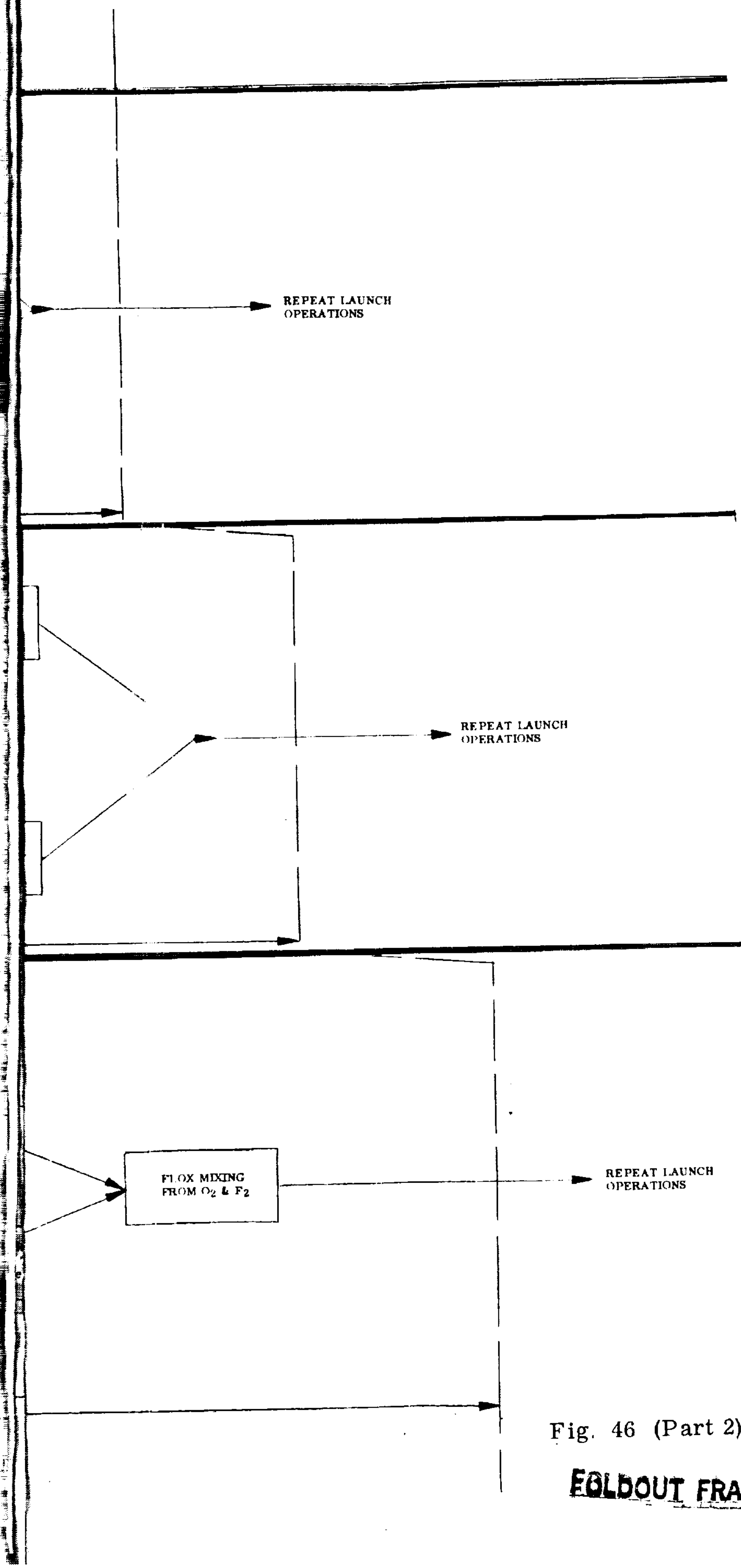

Fig. 46 (Part 2) Operations Flow Chart

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- The facility additions required for the propulsion system support and servicing are complete and operational, thus the analysis is representative of routine launch sequence.
- The facility systems are in a standby rather than a readiness condition. Propellant storage and transfer systems must be serviced, leak checked, and functionally checked prior to admitting propellants from ground transportation.
- Time periods on the chart are relative because vehicle electrical and electronic subsystems operations requirements were not a part of this study. The time periods shown are realistic for operations peculiar to the propulsion system operations and are consistant within the propellants studied.
- Operations on the chart are indexed at the point in the sequence where the launch pad is occupied by the vehicle. This was done to show the relationship between launch pad tie-up time and propellant combination.
- Recycle operations were analyzed on the basis that vehicle repair activity would not require demating of the vehicles, but that the propulsion system would be placed in an inert-safe state.

The following propulsion/propellant systems operational functions were considered:

- Facility and off-pad vehicle operations
  - Facility propellant systems functional checks, leak checks, and passivation
  - Facility storage tank propellant loading
  - Vehicle propulsion system leak checks, cleanliness, passivation
- On-pad vehicle propulsion checkout operations
  - Propulsion system functional sequence, leak checks
  - Vehicle tanks propellant preload conditioning
- Propellant loading and launch operations
  - Vehicle tanks propellant loading and topping
  - Countdown and liftoff
- Recycle (abort)
  - Vehicle tanks propellant draining

- Vehicle tanks safe inerting
- Hold for rework

Figure 46 shows the paths, task relationships, and task dependency of the various activities, the operational complexity, and the resulting relative time requirements. As expected, a vehicle using earth storable, N_2O_4/A-50, propellants is the least complex, and requires the shortest launch pad time and the shortest facility preparation time. This is due primarily to the minimal amount of propellant-facility and vehicle-tank preconditioning operations required.

The most time-consuming and complex operations occur with the vehicle using FLOX/CH_4. This longer time is due primarily to the operations required for mixing of liquid oxygen and liquid fluorine to obtain the FLOX, in addition to the necessary passivation operations for a fluorinated oxidizer. Procurement of FLOX in a premixed condition would eliminate this step.

A propulsion operations time comparison is shown in Table 26. The total on-pad time varies from 124 hours for N_2O_4/A-50 to 132 hours for FLOX/CH_4. Additional time required if a recycle is necessary is 12 hours for N_2O_4/A-50 and from 24 to 31 hours for the remaining propellants. The N_2O_4/A-50 system could therefore be recycled between one daily launch window and the next, while the remaining propellants will probably require two days.

3.2.6 Propellant Hazards Comparison

Each of the propellants under study presents some hazard to personnel. The least hazardous are H_2, O_2, and CH_4 which are all non-toxic and require no respiratory protection, but do require body protection to prevent frost type burns. All of the remaining propellants are quite toxic with threshold limits varying from 50 ppm for NH_2 for NH_2 to 0.05 ppm for OF_2, based on values recommended by the American Conference of Government Industrial Hygienists, 1968. Special breathing equipment is required. Toxicity of F_2 and FLOX is stated as 0.1 ppm, though recent studies indicate this may be raised to 1.0 ppm. All will damage body tissue through burns or dermatitis, and body protection is required.

A summary of toxicity and body tissue personnel hazards is presented in Table 27.

All of the propellants are potentially hazardous through fire, explosion, and stability characteristics. Each of the fuels will burn in the presence of air. B_2H_6 is the most easily ignited in air, with flammability limits between 0.9 and 93 percent by volume and auto ignition temperature of 300°F. This is followed closely by A-50 at 2.0 to 90 percent and 450°F. The least hazardous is CH_4 at 4.0 to 15 percent and 1200°F.

All of the fluorinated oxidizers are highly reactive with most substances at room temperature. All are stable. Table 28 summarizes fire, explosion, and stability characteristics.

3.2.7 Propellant Spill Disposal

Inadvertent spills are always a hazard, and must be handled in a manner that will keep danger to equipment and personnel to a minimum. Although preventative measures are taken to preclude such an occurrence, the possibility still exists. Accidental damage to equipment, contaminants in liquids, human error in operational procedures, etc., cannot be completely eliminated; therefore, the system must reduce spill hazard to a minimum.

Spills expose personnel and equipment to the dangers of explosions, fire, and toxic and /or corrosive liquids and vapors. It is imperative that personnel be trained in the handling and safety procedures for the materials in use. This alone, however, is sufficient to minimize the dangers involved. The equipment must be designed with built-in safety measures, including drain troughs, spill basins, water dilution, heat sinks, and chemical neutralizers.

Fluorine, FLOX, and OF_2 Spill Disposal. A drain trough to transport these propellants to a somewhat removed spill basin should be provided. The basin need only be removed from directly underneath the tower. This will allow corrosive vapors to rise without direct impingement on the vehicle and launch equipment. All basins and troughs should

Table 26

PROPULSION OPERATIONS TIME COMPARISON

		A	B	A + B	C	A+B+C
PROPELLANT	FAC/AGE AND OFF PAD VEHICLE OPS. (WKS)	ON-PAD PROPULSION CHECKOUT OPS. (HR)	PROPELLANT LOAD & LAUNCH (HR)	TOTAL ON-PAD (HR)	RECYCLE (HR)	TOTAL ON-PAD W/RECYCLE (HR)
N_2O_4/A-50	2	120	4	124	12	136
O_2/H_2	2.7	125	4.7	129.7	24	153.7
F_2/NH_3	2.9	122.5	4.7	127.2	29	156.2
OF_2/B_2H_6	3.0	125	4.7	129.7	26.5	156.2
F_2/H_2	2.9	125	5.1	130.1	29	159.1
FLOX/CH_4	3.2	127	5.0	132.0	31	163

Table 27

PERSONNEL HAZARDS COMPARISON

PROPELLANT	*TOXICITY (THRESHOLD LIMIT VALUE)	RESPIRATORY PROTECTION REQUIRED	EXPOSURE TO BODY TISSUE	BODY PROTECTION REQUIRED
B_2H_6	0.1 ppm/0.1 mg/m^3	Self cont. breathing equipment	Skin dermatitis	Gloves and face
H_2	None	None	Frost type burns	Gloves and face
CH_4	None	None	Frost type burns	Gloves and face
NH_3	50 ppm/35 mg/m^3	Canister mask or self-contained breathing equipment	Chemical burns	Gloves and face
A-50	1 ppm	Self contained breathing equipment	Chemical burns	Gloves and face
FLOX	0.1 ppm/0.2 mg/m^3	Self contained breathing equipment	Severe acid type burns	Full acid type suits
F_2	0.1 ppm/0.2 mg/m^3	Self contained breathing equipment	Severe acid type burns	Full acid type suits
OF_2	0.05 ppm/0.1 mg/m^3	Self contained breathing equipment	Acid type burns	Full acid type suits
O_2	None	None	Frost type burns	Gloves and face protector
N_2O_4	5.0 ppm/9 mg/m^3	Self contained breathing equipment	Acid type burns	Acid resistant gloves, boots and face

* Based on recommended values adopted by American Conference of Governmental Industrial Hygienists, 1968

Table 28

FIRE, EXPLOSIVE AND STABILITY HAZARDS COMPARISON

PROPELLANT	FLAMMABILITY LIMITS IN AIR % BY VOL	AUTO IGNITION TEMP. °F (IN AIR)	GENERAL COMMENTS
B_2H_6	0.9 to 93	300°F	Very low auto ign. temp. Contaminants can cause fires at room temp. Decomposes above -112°F.
H_2	4.0 to 74.2	1075	Fire potential if spills or leaks occur – (stable)
CH_4	4.0 to 15	1200	Fire potential if spills or leaks occur – (stable)
NH_3	16.1 to 26.8	1200	Fire potential if spills or leaks occur. Disassociates above 900°F.
A50	2.0 to 90	450	Fire potential if spills or leaks occur. Decomposes above 500°F.
FLOX	———	———	High rate of reaction with fuels, water. Reacts with most substances at room temp. (stable)
F_2	———	———	High rate of reaction with fuels, water. Reacts vigorously with most substances at room temp. (stable)
OF_2	———	———	Reacts with fuels. Reacts with most substances but to lesser degree than F_2. (stable)
O_2	———	———	High rate reaction with hydrocarbons (stable)
N_2O_4	———	———	Supports combustion in contact with fuels (stable)

be constructed of concrete. The spill basin can be lined with limestone for reaction. A water deluge system could be employed. A system of directional control gates may be necessary if the fluorine and fuel basins are incompatible. If a fire develops, the reaction is likely to be so rapid that no attempt can be made to extinguish the flame. After the fluorine-fed fire has subsided and the fluorine has been consumed, or has evaporated, efforts should be directed toward reducing secondary fires. Spills may be handled by remote application of water fog, fine water spray, or soda ash to promote smooth, rapid combustion of the fluorine. These problems and solutions also apply to FLOX and OF_2.

Hydrogen Spill Disposal. The existing spill disposal system will be more than adequate. Crushed rock should be used in the basin to increase the exposed surface area of the basin and its heat-sink capability. Hydrogen can be disposed of by vaporization, which will be accelerated by the increased heat sink. Hydrogen gas is extremely flammable, and a serious fire hazard always exists when hydrogen-gas vapors are in the area. With no impurities present, hydrogen burns in the air with an invisible flame. Extreme measures should be taken to prevent spark discharge. A hydrogen fire can be effectively controlled with heavy concentrations of water, CO_2, or steam.

Oxygen Spill Disposal. The existing spill system will be adequate. Disposal will take place by natural vaporization. Crushed rock will help accelerate vaporization. If a fire develops, all flow should be shut off. For large spill fires, wait until the oxygen has evaporated, and then use Class B fire extinguishing methods on remaining fires. Small spill fires may be extinguished directly using large quantities of water. The potential for an explosion is always present with spilled oxygen.

Methane Spill Disposal. Spill basin design for hydrogen is adequate. Ths spill should be deluged with water or water spray to reduce fire hazard. Fire hazard is not as great as with hydrogen. If fire does develop, the flame will be visible and can be extinquished with water, CO_2 , or steam.

Ammonia Spill Disposal. The existing spill basin design is adequate. Since this fuel will be used with fluorine, a directional control gate may be necessary to separate spill basins (refer to discussion on fluorine) if two basins are required. Water deluge is required to reduce fire, explosion and toxic hazards.

The flammability range of ammonia is at higher concentration than for hydrocarbons, but large spills will present a fire hazard. Ammonia fires are very difficult to extinguish. Water fog is recommended for ammonia fires because it cools the burning surfaces and reduces the vapor pressure by absorption and dilution. large quantities are required. The explosion hazard of ammonia is relatively low compared to hydrogen.

N_2O_4 Spill Disposal. The existing spill basin is adequate. The areas should be deluged with water to reduce the fire hazard; however, water will accelerate fuming. Nitrogen tetroxide supports combustion; if fire is present, deluge with water. Continued application of large quantities of water will eventually dilute the oxidizer so that combustion is no longer supported. Remaining airsupported fires may be extinguished by ordinary means.

Aerozine-50 Spill Disposal. Use present spill basin. Area should be deluged with water to reduce the fire hazard. If fire is present, water is the safest and most effective agent to use. Only water is recommended for oxidizer-supported fires if it is compatible with the oxidizer. If the fire is air-supported and it is a small spill, bicarbonate-base (powder-type) agents are the most effective. Water fog or carbon dioxide may also be used. If the spill is large and air-supported, only large amounts of coarse spray water are recommended. The water fog, CO_2, and bicarbonate methods are subject to backflashes and explosive reignitions. The A-50 propellant readily forms an explosive mixture with air which can be ignited by a spark or flame.

B_2H_6 Spill Disposal. A drain trough similar to the OF_2 system should be provided. This will carry away large quantities of liquid and will minimize potential damage to the vehicle and launch facilities in the event of a fire. If a fire develops, it should be controlled with a water deluge, sprinkler, or fog system. Combustion in air is readily detectable by a green-organge to blue flame. The products of incomplete combustion that will be present are even more toxic than diborane vapors. All disposal equipment should be operated remotely.

3.2.8 Vent Gas Disposal

Vent gases must be disposed of for two primary reasons: to reduce the potential for fire or explosion and to eliminate the toxic and corrosive dangers. Vent gases are usually routed through a pipe to an area remote from the launch vehicle and personnel. It is then free-vented to the air or burned.

Fluorine, FLOX, and OF_2 Gas Disposal. These gases are extremely toxic and can cause severe burns and pulmonary edema. The total mass of vented vapors should be kept to an absolute minimum. All gas should be piped to a remote vapor disposal unit. This unit may contain charcoal to reduce the fluorine, FLOX, or OF_2 content sufficiently if small quantities are vented. Fluorine gas may also be combined with propane during a burning process. Hydrogen fluoride gas will be a by-product of combustion. This gas is also toxic and may be scrubbed through charcoal. The latter process may be more convenient since less charcoal is required. The container for charcoal need only be an open concrete pit that free-vents the gases to the atmosphere Periodic replacement of the charcoal is necessary since it will be consumed combustion.

Hydrogen Vent Gas Disposal. Hydrogen vapor in the quantities used during loading can be free-vented to the atmosphere through a remote standpipe. It can be burned if necessary in the burn pond provided for the S-IVB.

Oxygen Vent Gas Disposal. Oxygen can be free-vented to the atmosphere.

CH_4, NH_3, N_2O_4 Vent Gas Disposal. Vapors can be free-vented to the atmosphere in quantities formed during loading of spacecraft or run through a vapor disposal unit such as provided for the Apollo LM.

B_2H_6 Vent Gas Disposal. Diborane vapor can safely be disposed of by piping to a remote area where it can be burned. Complete combustion should be effected if possible such as in a burn pond. The products of complete combustion are not toxic and can be freely vented to the atmosphere. The products of incomplete combustion are more toxic than straight diborone vapors but should be present in sufficiently small quantities so as not to present a hazard.

3.3 GROUND FACILITIES

The facilities required to perform the required ground operations were identified, with emphasis on showing the capabilities of existing facilities together with requirements for new facilities. Propellant facility elements considered include:

- Storage tanks
- Transfer lines
- Loading flow control system
- Vehicle tank preloading conditioning systems
- Vehicle tank preloading safe verification systems
- Propellant safe-vent systems
- Vehicle tank propellant no-vent systems

3.3.1 Basic ITL Capabilities

The Titan III vehicle is launched from Integrated Test and Launch (ITL) Complex 40 and 41. This facility utilizes the ITL concept wherein the integration and checkout of complete vehicles, including spacecraft, is accomplished in a central area, the assembled vehicle is transferred to the launch pad, and the vehicle is launched. This facility presently provides for receiving, transportation, storage, assembly, maintenance, checkout, and launch of the SLV-5. The basic facility comprises the Vertical Integration Building (VIB), the Solid Motor Assembly Building (SMAB), and two launch complexes, connecting transporter trackage, and the vehicle transporter. The basic ITL area plan view was presented earlier in Fig. 35. An aerial view is shown in Fig. 47.

3.3.1.1. Vertical Integration Building (VIB)

An aerial view of the VIB is presented in Fig. 48. The functions performed in the VIB are assembly and checkout of the core vehicle stages and payloads, and launch control of the complete vehicle when it is on the launch pad. It contains four assembly and checkout bays, support shops, quality control labs, a launch control center, an AGE van area, and a receiving area. The high bay area containing the four assembly and checkout bays is approximately 100 ft wide by 350 ft long and 215 ft high. A 20-ton bridge crane with a 180-ft lift services this area. Eleven service platforms between the 35 and 155 ft levels are provided, the top five (from 112 to 158 ft) have provisions for continuous vertical adjustment and are also retractable from the fully extended payload access position. The launch control center is located on the second floor over the AGE van area and contai ns approximately 14,000 sq ft. Each of the payload control rooms contains approximately 550 sq ft and the vehicle control room contains 900 sq ft. The following services are supplied:

1. Air Conditioning: Office, shop, and launch control areas only (vehicle assembly area not controlled)
2. GN_2 Storage: 275 cu ft at 5500 psi
3. LN_2 Storage: 28,000 gal. at 50 psi
4. He: Two trailers at 2400 psi
5. 120/208/460 volt power, ac

Telephone, instrumentation landline, and data transmission systems exist between the LCC and the launch pad. Thses have sufficient capacity to handle the needs of the programs now supported by this complex.

3.3.1.2 Solid Motor Assembly Building (SMAB)

The SMAB is a high-bay building, located between the VIB and the launch pads, which is used to assemble the segmented solid motors and mate them to the core vehicle which has already been assembled and transported from the VIB.

Fig. 47 Integrate Transfer Launch Area, Aerial View

Fig. 48 Vertical Integration Building, Aerial View

3.3.1.3 Launch Complexes 40 and 41

A plan view of Launch Complexes 40 and 41 is presented in Fig. 49. The launch complex is an area 1100 ft in diameter which includes the launch pad structure with exhaust duct, mobile service tower, umbilical tower, AGE building, air conditioning shelter, protective clothing building, gas storage area, guidance buildings, fuel holding area, oxidizer holding area, ready building, and the complex support building. Launch complexes 40 and 41 are nearly identical.

3.3.1.4 Launch Pad Structure

The launch pad structure contains the foundation for supporting the transporter and vehicle and the exhaust duct for deflecting the exhaust gases away from the facility. Exhaust deflectors are the "dry" type.

3.3.1.5 Mobile Service Tower (MST)

The mobile service tower is a steel-framed rigid structure mounted on self-propelled wheeled trucks riding on two standard gage railroad tracks. The tower is 240 ft high and 121 ft wide. Service platforms are provided at 13 levels, the top four being adjustable vertically. All platforms have retractable sections. A bridge crane with 50-ton and 10-ton hoists with a 180 ft hook height is mounted on the structure. A 30 by 40 ft environmental enclosure is provided between elevations 216 and 263 ft. The service tower is rolled away 585 feet from the vehicle prior to launch.

3.3.1.6 Umbilical Tower

The umbilical tower is a steel frame structure anchored to the launch pad. It is 170 ft high and approximately 50 ft wide. Service platforms are provided at 13 levels; the top four are adjustable vertically. All platforms have retractable sections. On the tower are located the flexible service and launch pull-away lines and disconnecting equipment required between the tower, the umbilical mast, and the vehicle. The following service lines are mounted on the tower:

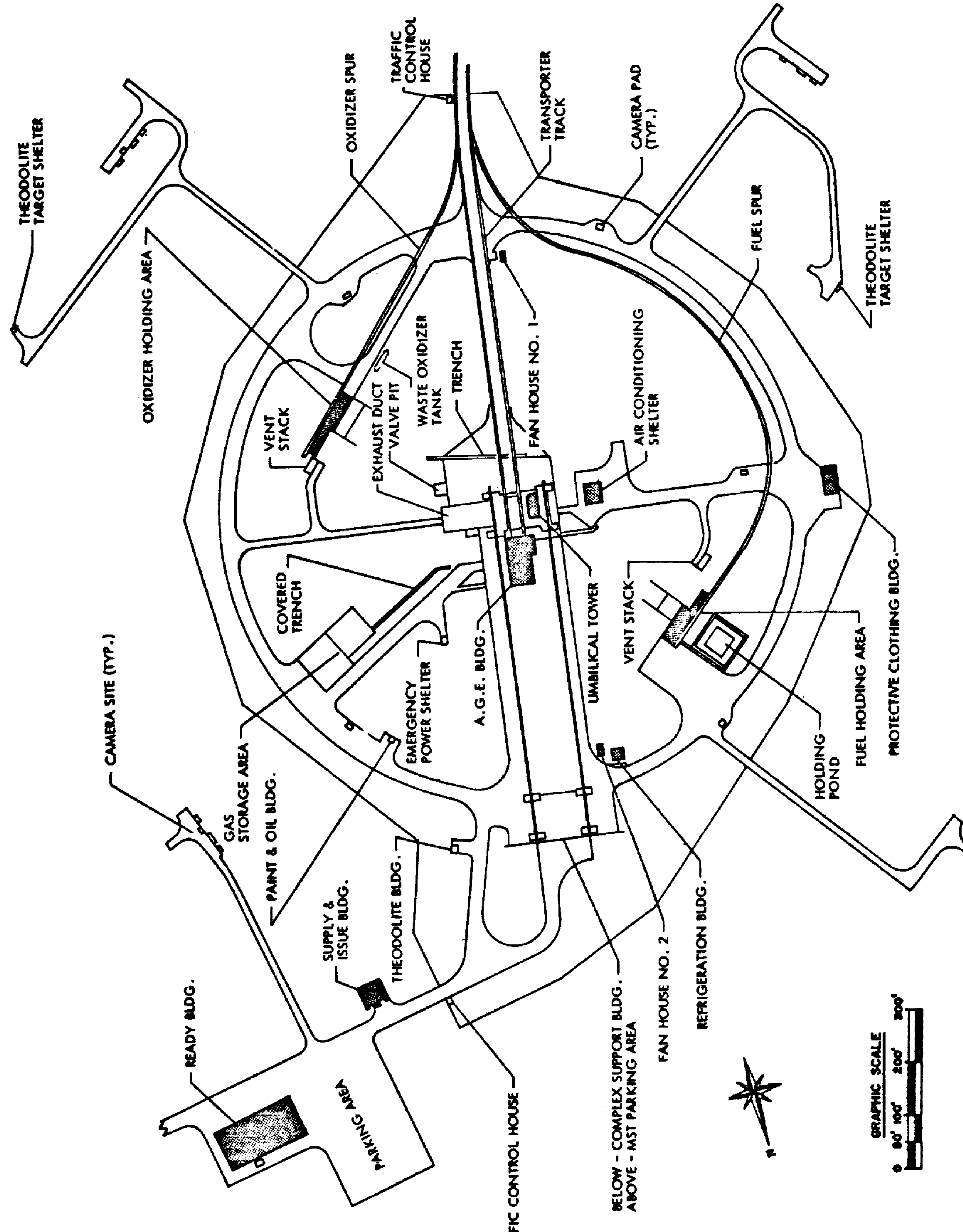


Fig. 49 Launch Complexes 40 and 41, Plan View

1. Helium fill lines: 3500 psig and 150 psig
2. Oxidizer (N_2O_4) return, fill, drain, vent, and waste lines
3. Fuel (Hydrazine) return, fill, drain, vent, and waste lines
4. Vehicle air conditioning duct
5. Guidance, cooling, and return lines
6. Water service

3.3.1.7 AGE Building

The AGE building is a two story concrete structure located between the MST tracks adjacent to the launch pad. The upper floor is level with the transporter tracks and houses the AGE vans (two core AGE vans and two payload AGE vans). The lower level contains AGE racks such as communications, power supplies, GN_2 controllers, and timing equipment.

3.3.1.8 Complex Support Building

The complex support building is a 2500 sq ft, one story, concrete building located below the MST in its stowed location. It provides storage and shop space for the vehicle and payload contractors.

3.3.1.9 Vehicle Transporter

The vehicle transporter consists of the diesel electric locomotives, undercarriage assembly, launch platform, thrust mounts, umbilical mast, cable support, and AGE vans. The undercarriage consists of the tracks and platform elevating system used for placing the launch platform on the facility tiedowns. The launch platform supports the thrust mount for the Titan IIIA "core only" configuration or the thrust mount for the Titan IIIC when the solid motors are used. The thrust mounts have the capability of leveling and aligning the vehicle. The umbilical mast provides mounting and support for the electrical umbilicals and cabling to the vehicle during checkout, transport, and launch. The AGE vans (core and payload) provide protection and space for the electronic AGE used during checkout and launch. The vans are capable of being towed either on roadways or standard gage railroad tracks.

3.3.1.10 Propellant Holding Area

In the propellant holding area storage tanks for 28,000 gal of oxidizer (N_2O_4) and 22,000 gal of fuel (Hydrazine – UDMH) are provided. The fuel and oxidizer facilities are placed on opposite sides of the launch pad. Transfer is handled by local control at the AGE transfer set.

3.3.1.11 Gas Storage Area

In the gas storage areas, storage is provided for the following:

1. LN_2 – 28,000 gal fixed tank and 1800 gal trailer
2. GN_2 – 1800 cu ft (water volume) at 5500 psig on Pad 40 and 1650 cu ft at 5500 psig on Pad 41
3. Helium – 300 cu ft at 5500 psig compressed from 2400 psig trailers

3.3.1.12 Air Conditioning

The following air conditioning is available to the vehicle at both the VIB and the launch pad:

- Flow Rate – 70^{+10}_{-5} lb per min
- Pressure – 20 ± 5 in. water
- Temperature – 52.5° to 62.5°F ±3°F
- Filtration – 25μ nominal

3.3.2 ITL Modifications Recommended for Adding Centaur and New Propellants

Modifications required to the ITL to support Titan/Centaur launches are described in detail in the Titan IIID/Centaur Integration document MCR-69-1 prepared by Martin-Marietta Corporation. New storage areas for oxidizer, fuel, and gas are shown on Fig. 50 as reproduced from MCR-69-1 (Vol. II).

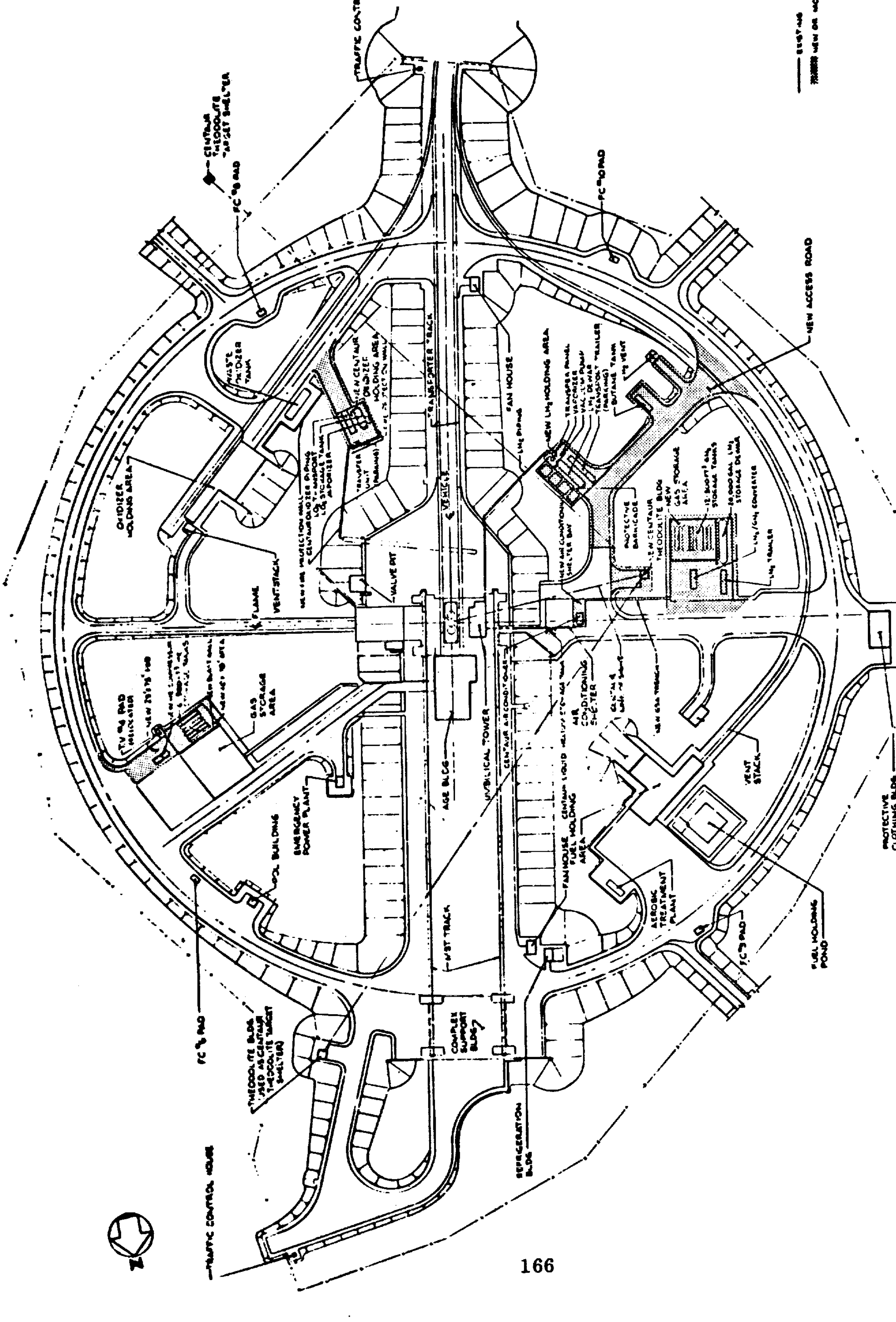


Fig. 50 Titan III/Centaur Launch Complex

Propellant storage requirements for a pad-loaded Mars Orbiter are shown in Table 29 by propellant type. These requirements are then translated into recommendations for new or modified storage facilities and vent disposal at Complex 40 and 41, and are presented in Table 30 together with a code identifying specific locations at the Complex. These locations may be found by referring to the corresponding coded location on Fig. 51.

3.3.3 Basic Propellant Loading Systems

Basic loading systems are required for each of the candidate propellants. This system will be essentially identical whether loading takes place on the launch pad or in a remote facility. Loading system schematics are presented in Figs. 52 through 58 for N_2O_4, A-50, O_2, H_2, CH_4, F_2, OF_2, FLOX and B_2H_6.

The simplest loading system is that for the earth storables, N_2O_4 and A-50, shown in Fig. 52. The basic requirements are for GN_2 pretransfer purge of the transfer lines and vehicle tanks, fill rate control valves, propellant level sensing instrumentation, vent disposal, and a quick disconnect between transfer lines and vehicle.

Loading of liquid O_2 is complicated only by the need for vacuum jacketed storage tanks and foam insulated transfer lines as shown in Fig. 53. Since liquid oxygen is relatively inexpensive and venting direct to the atmosphere is quite acceptable a refrigeration system is not required. Pretransfer purge is not required.

The basic loading system for liquid H_2 shown in Fig. 54 adds insulation, vacuum jacketed storage tanks and transfer lines, remote-located elevated vent stack, and condensible/reactant sensing instrumentation.

The loading system for CH_4 and NH_3 shown in Fig. 55 requires vacuum insulated transfer lines and a remote-located, elevated vent stack. Active refrigeration is not required since liquid CH_4 and NH_3 are relatively inexpensive and venting direct to the atmosphere is acceptable.

Table 29

PROPELLANT PAD STORAGE REQUIREMENTS

Propellant	Total Prop. (lb)	O/F	Ind. Prop. (lb)	Loading Factor	Minimum Pad Storage Quan (lb)	Recommended Pad Storage (lb)	Remarks (See Fig. 51 and Table 31)
F_2	3500	12	3230	1.1	3550	3800	Add storage
H_2			270	3.0	810	900	Use Centaur holding area
O_2	3900	6	3343	2.0	6686	7000	Use Centaur holding area
H_2			557	3.0	1671	1700	Use Centaur holding area
FLOX F_2	4050	5.25	(3400) 2810	1.1	3090	3300	Add storage
FLOX O_2			590	2.0	1180	1300	Use existing Centaur O_2 storage
CH_4			650	1.5	975	1100	Add storage
F_2	4050	3.3	3010	1.1	3310	3500	Add storage
NH_3			940	1.5	1410	1600	Add storage
OF_2	4100	3.0	3075	1.1	3380	3500	Add storage
B_2H_6			1025	1.1	1125	1300	Add storage
N_2O_4	5150	1.6	3170	1.05	3330	3500	Use existing Titan storage
A-50			1980	1.05	2080	2200	Use existing Titan storage

Table 30

MODIFICATIONS TO COMPLEX 40–41 FOR NEW PROPELLANTS

Propellant	Storage Location (See Fig. 51)	Increase Existing Capacity To (lb)	Add New Storage Capacity (lb)	New Vent Disposal Location (See Fig. 51)
F_2	D	–	3,800	F
H_2	A	17,500	–	–
O_2	C	102,000	–	–
H_2	A	18,300	–	–
FLOX (F_2)	D	–	3,300	F
(O_2)	C	46,300	–	
CH_4	B	–	1,100	–
F_2	D	–	3,500	F
NH_3	B	–	1,600	–
OF_2	D	–	3,500	F
B_2H_6	B	–	1,300	G
N_2O_4	E	No Increase	–	–
A-50	H	No Increase	–	–

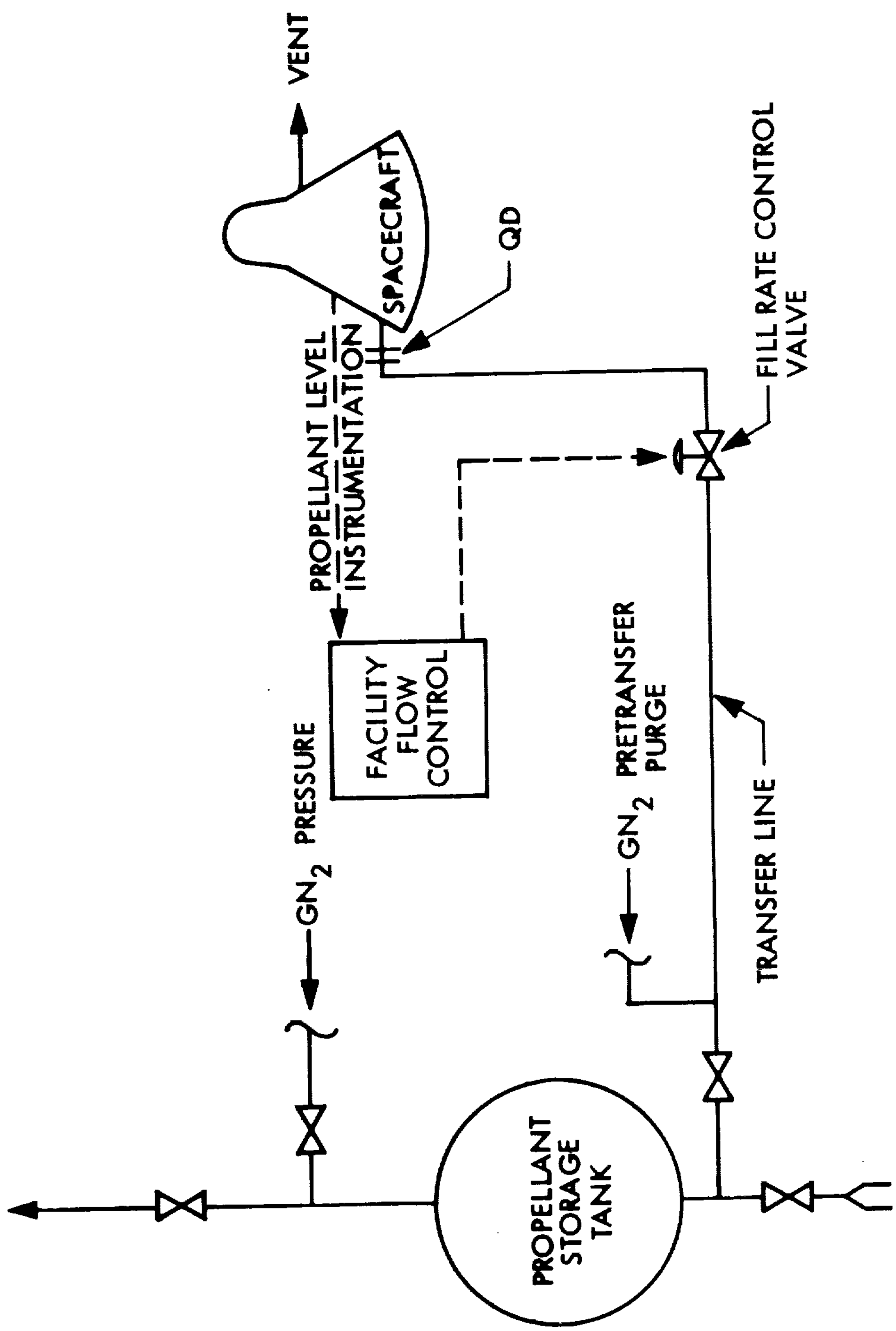
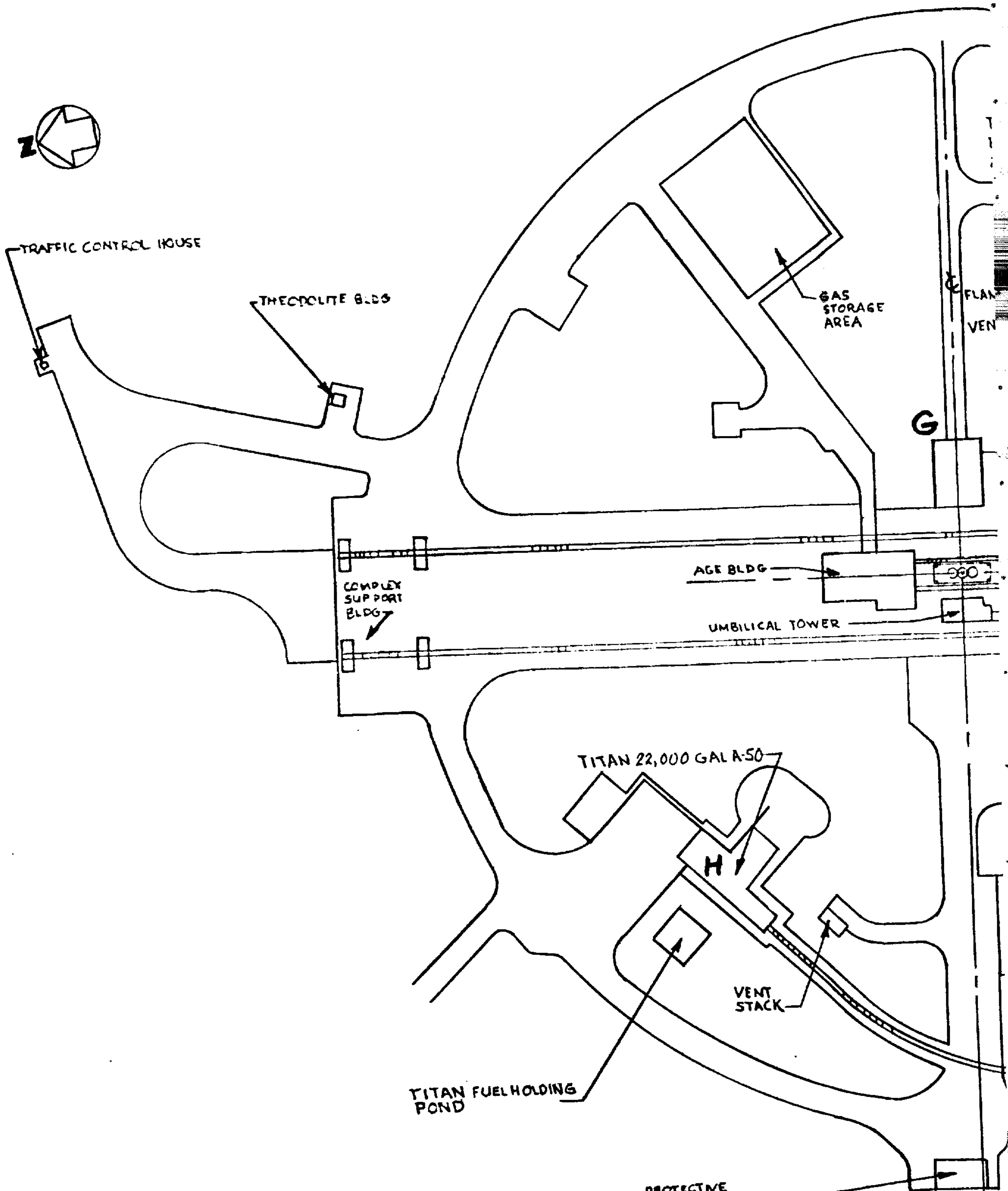


Fig. 52 N_2O_4 or A-50 Basic Propellant Loading System



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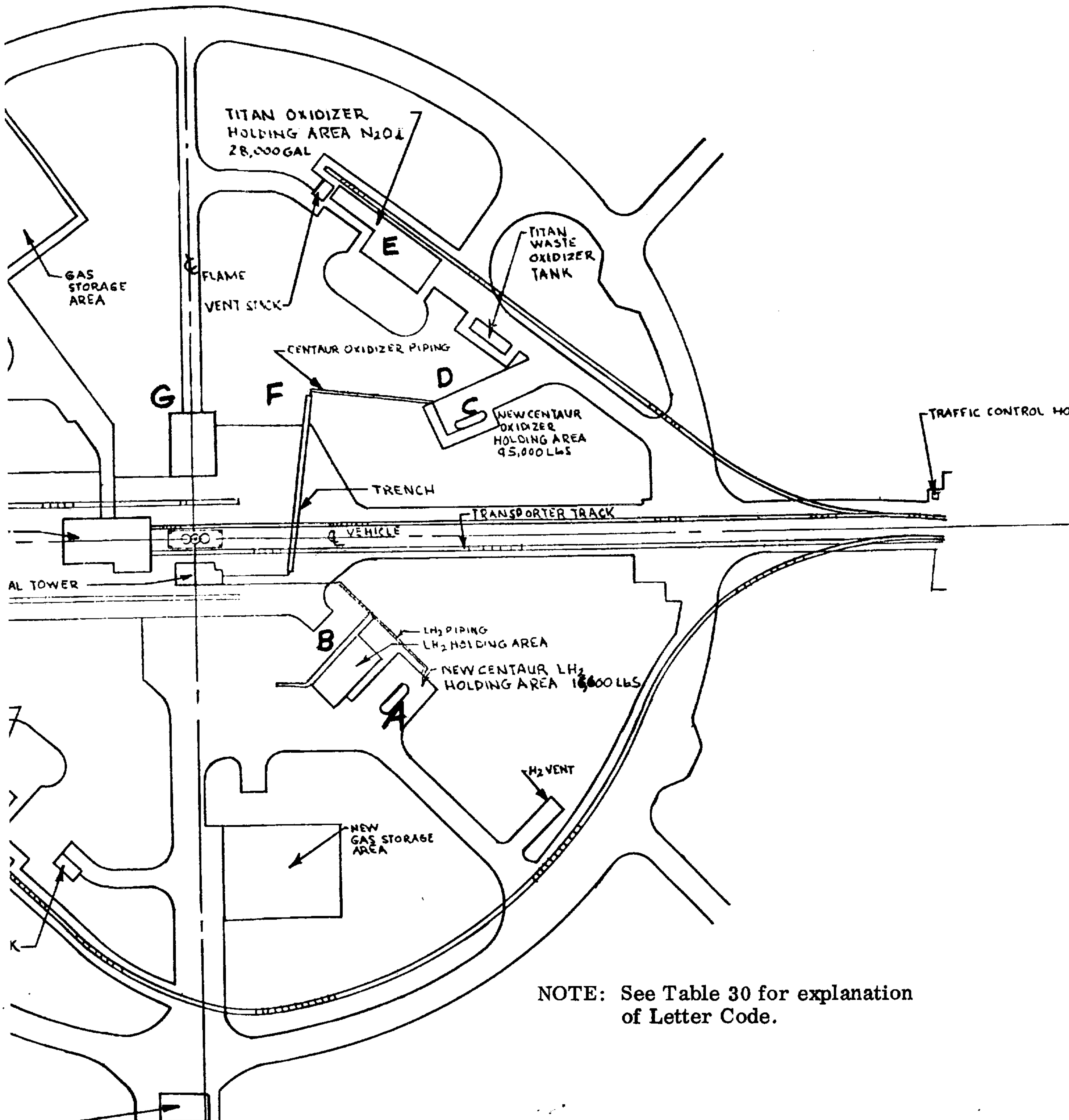

NOTE: See Table 30 for explanation of Letter Code.

FOLDOUT FRAME 2

Fig. 51 ITL Complex 40 – 41 Modified for Titan III/Centaur

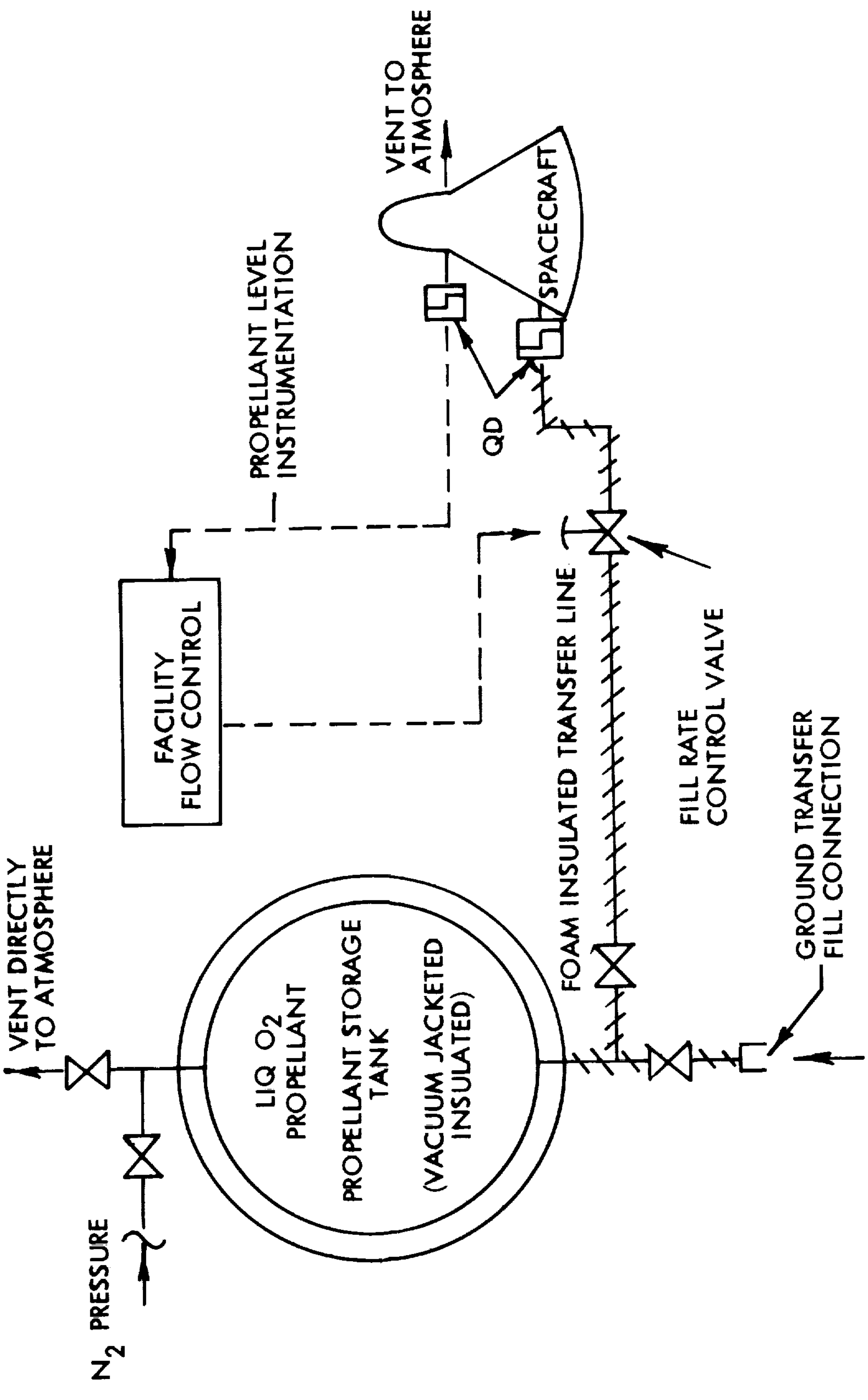

Fig. 53 O_2 Basic Propellant Loading System

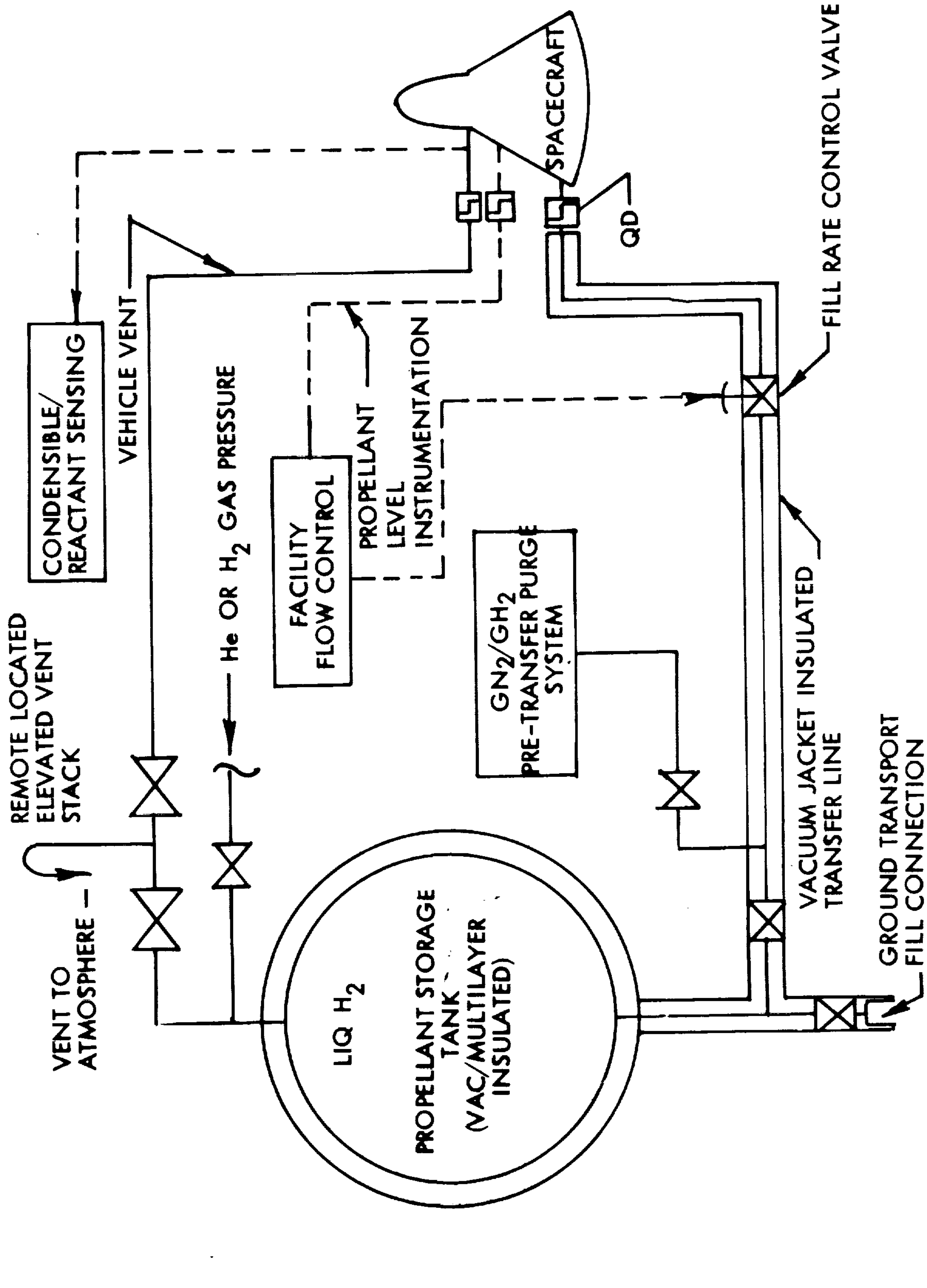

Fig. 54 H_2 Basic Propellant Loading System

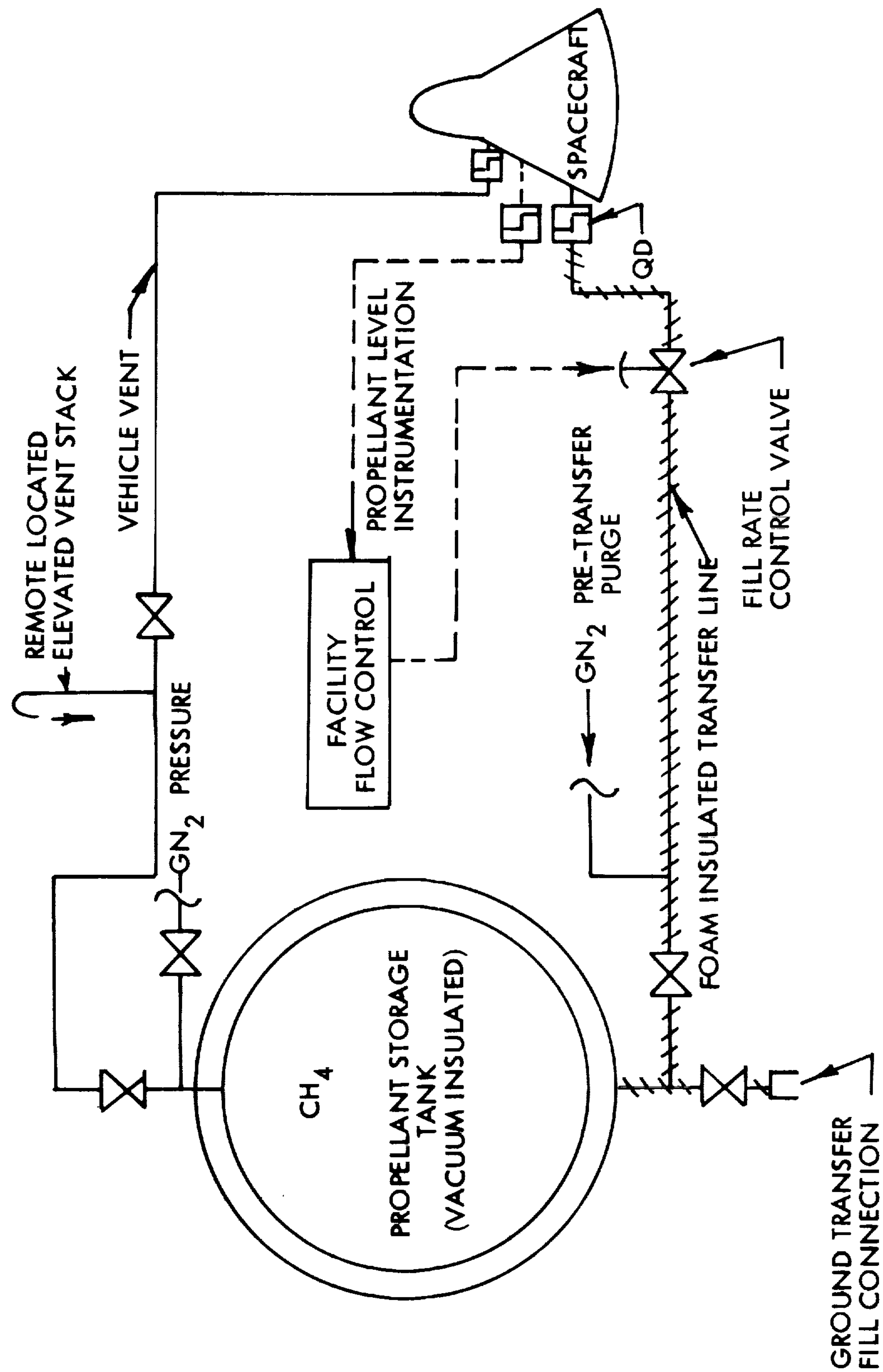

Fig. 55 CH_4 or NH_3 Basic Propellant Loading System

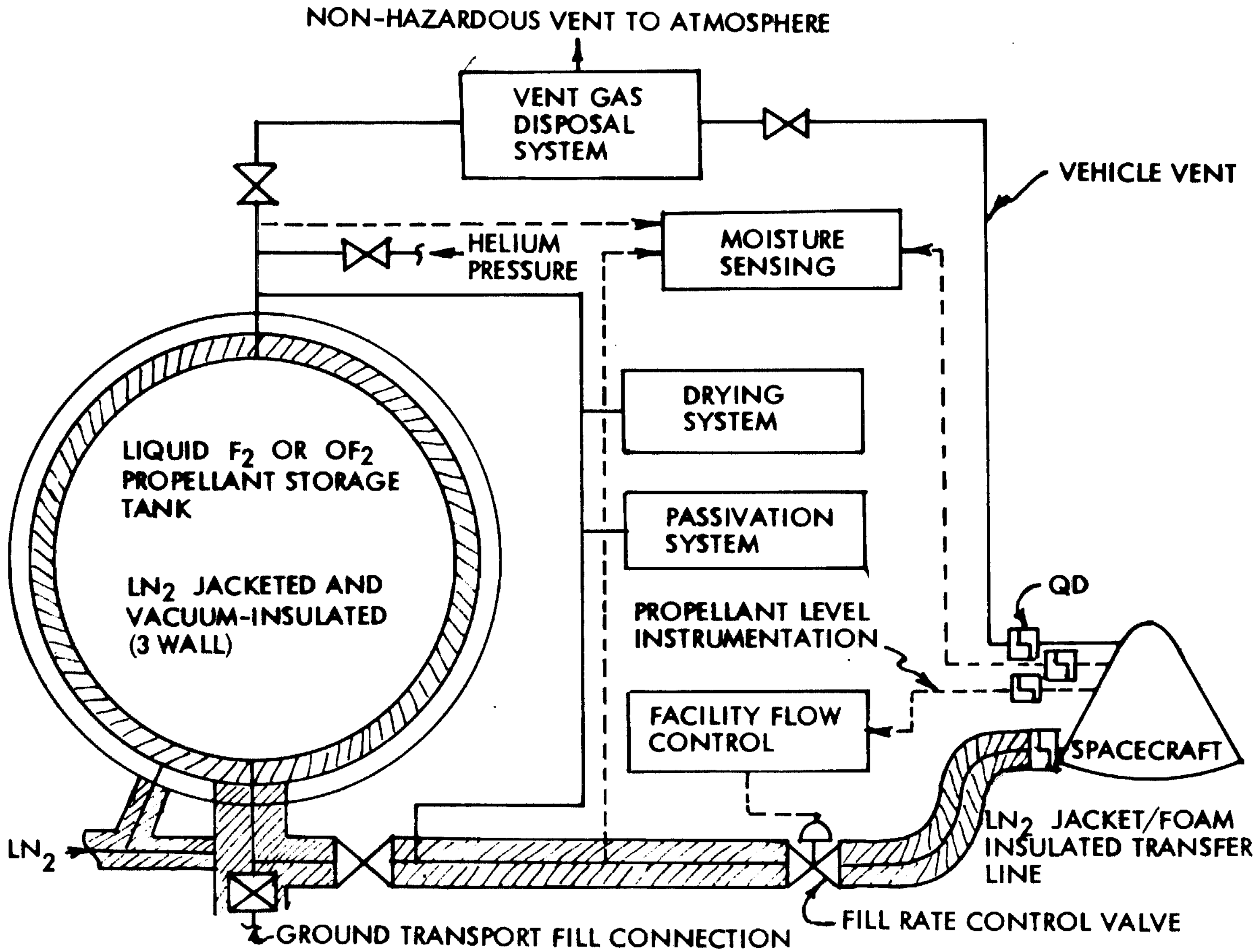

Fig. 56 F_2 or OF_2 Basic Propellant Loading System

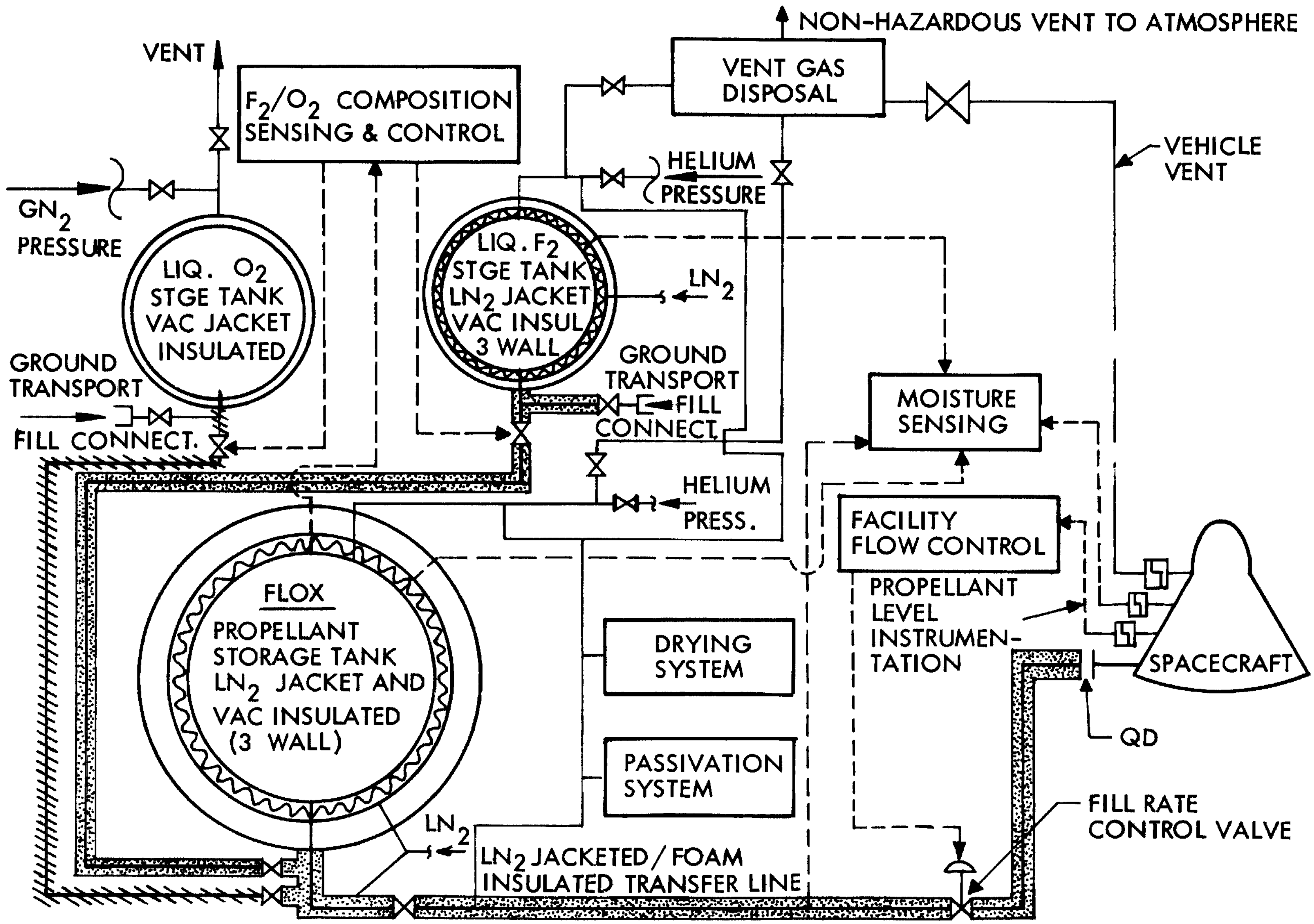

Fig. 57 FLOX Basic Propellant Loading System

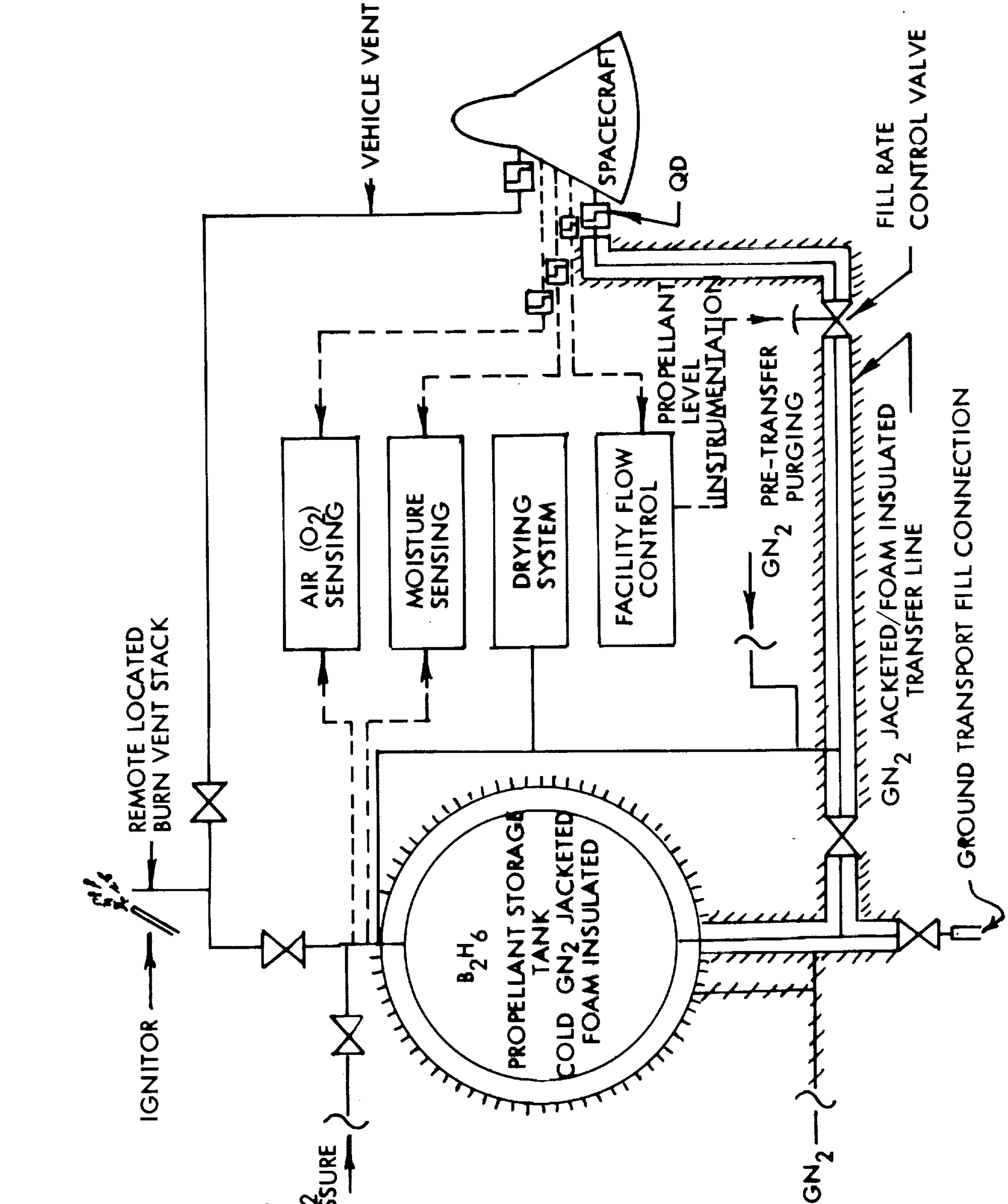

Fig. 58 B_2H_6 Basic Propellant Loading System

Loading systems for the fluorinated oxidizers become considerably more complicated. Since these oxidizers are quite toxic and reactive, and are relatively expensive, liquid N_2 jacketed and vacuum insulated storage tanks and liquid N_2 jacketed and foam insulated transfer lines are recommended. No venting should occur except to bleed off pressure following pressure transfer of the propellants. Vent gases are passed through a carbon scrubber disposal system before reaching the atmosphere. Figure 56 shows these features for a propellant loading system designed for liquid fluorine or liquid OF_2. Also shown for these propellants are additions of a drying system, a passivation system, and moisture sensing instrumentation.

The most complex system is that shown in Fig. 57 for FLOX, including facilities for storing and mixing the liquid oxygen and liquid fluorine. Here all the elements of the separate O_2 and F_2 loading systems are required plus the vacuum insulated LN_2 cooled FLOX tank and FLOX mixture composition sensing and control.

The loading system for B_2H_6 is shown in Fig. 58. Cold gaseous N_2 cooling of the storage tanks and transfer lines is recommended, together with foam insulation to reduce the flow requirements for GN_2. Air and moisture sensing instrumentation and a drying system are required. Venting will be required only to bleed down pressure after propellant transfer. The vent gases are burned in a remotely located stack for disposal since B_2H_6 vapor is quite reactive with normal atmosphere.

3.3.4 Zero Vent Facilities

Zero vent facilities were discussed earlier under Section 3.2.2, with concepts for non-vented ground hold thermal conditioning systems shown in Figures 40, 41, and 42. These systems must all be mobile, staying with the propulsion stage from the time of propellant loading in a remote facility until immediately prior to launch.

3.3.5 Basic Design Considerations Vs. Propellants

Several basic design considerations influenced by propellant choice are:

- Propellant/materials compatibility
- Propellant liquid temperature range
- Propellant facilities flexibility
- Ease of on-pad component replacement
- Relative complexity of operational tasks and facility elements

All of these areas were examined and are discussed in turn in the following paragraphs.

3.3.5.1 Propellant/Materials Compatibility

Suitable materials are available for storage tanks, transfer lines, valves, seals, etc., for use with each of the candidate propellants. Passivation of all equipment is required for most of the propellants. Table 31 summarizes propellant/materials compatibility for each oxidizer and fuel for representative metals and non-metals. The compatibility of aluminum with B_2H_6 has not been definitely established, but preliminary results of a current investigation at Stanford Research Institute under JPL contract 951584 are favorable.

3.3.5.2 Propellant Liquid Temperature Range

The liquid temperature range at various pressures is shown in Fig. 59. This figure also shows that N_2O_4 and A-50 can be held on the ground indefinitely at ambient temperature, that all remaining propellants except H_2 can be cooled and maintained by liquid N_2, and the H_2 can be maintained by liquid He. All of the oxidizer/fuel combinations except F_2/H_2 and FLOX/CH_4 have overlapping liquid ranges at one atmosphere pressure. The FLOX/CH_4 liquid range can be made to overlap by pressurizing the FLOX to slightly over 100 psia.

3.3.5.3 Propellant Facilities Flexibility

An analysis was made from the standpoint of materials compatibility and liquid temperature design limitations to see how flexible a system might be in accommodating

Table 31

PROPELLANT/MATERIALS COMPATIBILITY

		METALS							NON-METALS				
		TITAN-IUM	MILD STEEL	STAIN-LESS STEEL	ALUMI-NUM	NICKEL	COPPER	BRASS	POLY-ETHYLENE	ASBESTOS	TEFLON	KEL-F	MYLAR
OXIDIZER	FLOX	NO	NO	304L	YES	YES	YES	YES	NO	NO	YES	YES	NO
	F_2	↓	↓	304L	↓	↓	↓	YES	↓	↓	↓	↓	↓
	OF_2	↓	↓	YES	↓	↓	↓	YES	↓	↓	↓	↓	↓
	O_2	NO	NO	YES	↓	↓	YES	NO	NO	↓	↓	↓	↓
	N_2O_4	YES	YES	YES	YES	YES	NO	NO	YES	NO	YES	YES	NO
FUEL	B_2H_6	*	NO	YES	*	YES	NO	YES	YES	YES	YES	YES	YES
	H_2	YES	↓	↓	YES	YES	YES	YES	NO	NO	↓	↓	YES
	CH_4	*	↓	↓	YES	YES	↓	YES	YES	YES	↓	↓	YES
	NH_3	YES	↓	↓	NO	NO	↓	NO	YES	YES	↓	↓	NO
	A-50	YES	NO	YES	YES	NO	YES	NO	YES	NO	YES	YES	YES

*INSUFFICIENT DATA AVAILABLE

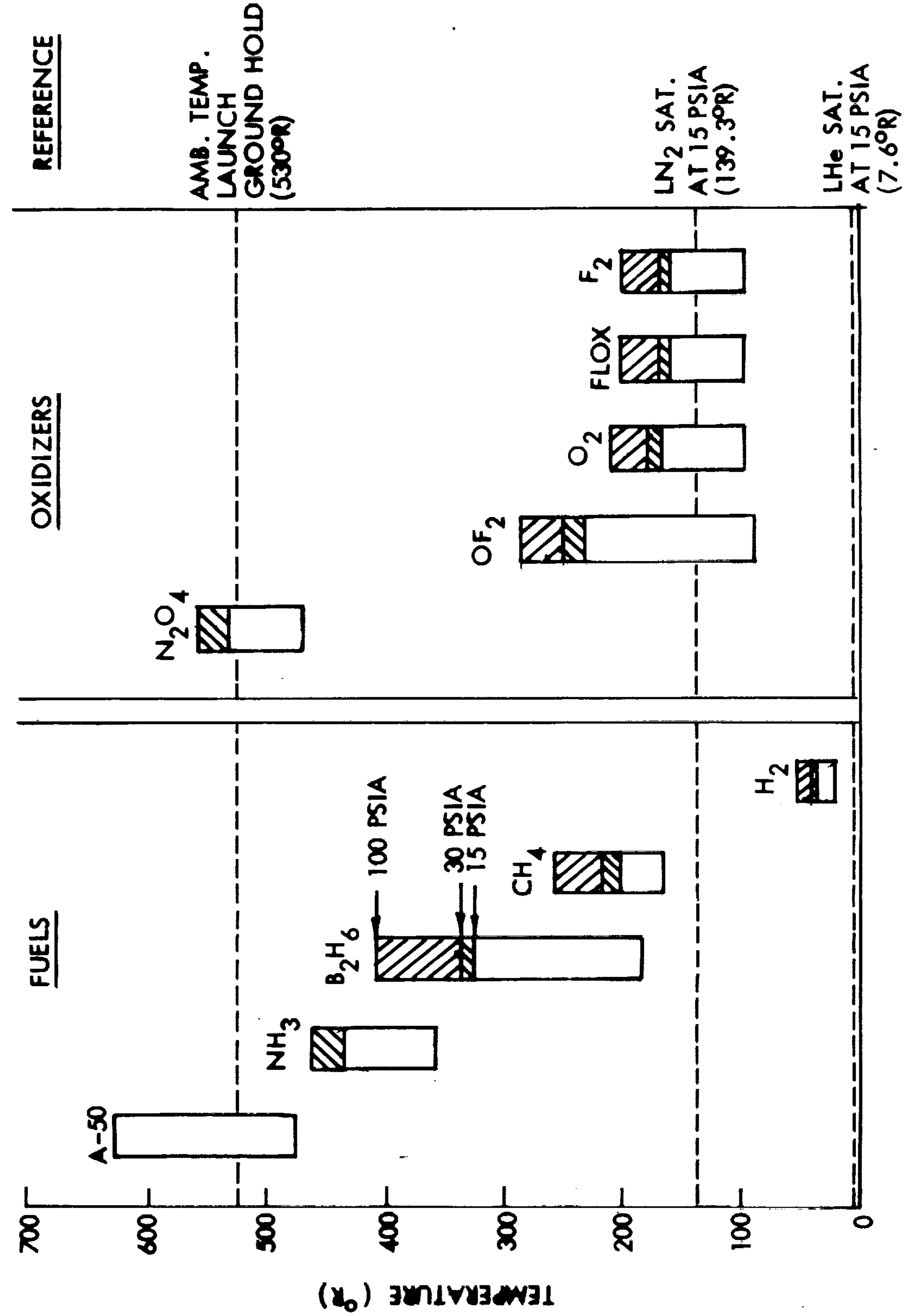

Fig. 59 Propellant Liquid Temperature Range

alternate propellants. The results are summarized in Table 32 and show that a system designed for FLOX or F_2 oxidizer could be used for OF_2, O_2, or N_2O_4 while a system design for H_2 or B_2H_6 would be suitable for CH_4, NH_3, or A-50.

3.3.5.4 On-pad Component Replacement

Servicing, checkout, inspection, and replacement of selected propulsion system components at the luanch pad is a key operational consideration, and any requirements in the area must be factored into the vehicle systems design criteria. The vehicle design must provide satisfactory location and accessibility of the selected components. Chosen components must meet both mission and accessibility requirements.

Replacement of a component that is not wetted with propellant and does not open the propellant system poses no new problems over existing systems.

Design of propellant distribution piping and plumbing, when wetted component removal is considered, is particularly sensitive to propellant choice. Highly reactive or toxic propellants require that the component and piping system be isolated and inerted prior to removal. Because of this inerting operation, normally by purging, complex piping geometry with many instrumentation taps, tubing, bands, and ties, or trapped spaces must be avoided.

Propellant system cleanliness is a mandatory consideration of all propellants under study. Certain propellants have particular contaminant problems which must be factored into vehicle design in order that decontamination can be accomplished or cleanliness maintained during component removal and replacement.

Table 33 identifies considerations to be factored into vehicle design as a function of propellant. This table was developed from analysis of hazards and compatibility data for the various propellants.

Table 32

PROPELLANT FACILITIES FLEXIBILITY

	OXIDIZERS					FUELS				
FACILITY DESIGNED AND BUILT FOR →	FLOX	F_2	OF_2	O_2	N_2O_4	H_2	B_2H_6	CH_4	NH_3	A-50
CAN ALSO BE USED FOR*	F_2	**	O_2	—	—	CH_4	CH_4	NH_3	A-50	—
	OF_2	OF_2	N_2O_4			NH_3	NH_3	A-50		
	O_2	O_2				A-50	A-50			
	N_2O_4	N_2O_4								

*ANALYSIS BASED PRIMARILY ON COMPATIBILITY AND LIQUID TEMPERATURE DESIGN LIMITATIONS OF SYSTEMS.

**FLOX ELIMINATED IN THIS COLUMN DUE TO NEED FOR LAUNCH SITE O_2, F_2 MIXING CAPABILITY REQUIREMENT.

Table 33

ON-PAD COMPONENT REPLACEMENT REQUIREMENTS WITH PROPELLANTS DRAINED

Propellant	Contaminant Control Requirements	Inerting Requirements (After Propellant Exposure)	Other Considerations
F_2	Particle, moisture, other reactive materials especially hydrocarbons	Drain, warm, purge	Passivated surfaces must be maintained or repassivated
FLOX	Same as F_2	Same as F_2	Same as F_2
OF_2	Same as F_2 (less reactive)	Same as F_2	Same as F_2 but less sensitive to disturbed passivated surfaces
O_2	Particle, hydrocarbons	Drain, warm	–
N_2O_4	Particle, moisture slightly, hydrocarbons	Drain, purge	–
H_2	Particle	Drain, warm, purge	–
CH_4	Particle	Drain, warm, purge	–
NH_3	Particle	Drain, purge	–
B_2H_6	Particle, moisture, air	Drain, warm, purge	–
A-50	Particle, moisture	Drain, warm	–

3.4 OPERATIONS AND FACILITIES COMPLEXITY

A summary of the factors that affect the ground operational requirements was made and the relative complexity of operational tasks and facility elements for alternate systems evaluated. The total number of ground operational tasks and the total number of facility elements are presented in Table 34 for each propellant combination as one measure of complexity. In preparing this table no attempt was made to compare complexity or reliability of one element to another, or to weight the difficulty of carrying out one operational task as compared to any other.

From Table 34 it is evident that the earth-storable combination N_2O_4/A-50 requires the fewest operational tasks and facility elements at 21 and 10, respectively. For the remaining propellants, the number of operations tasks range from 30 for O_2/H_2 to 38 for FLOX/CH_4 and the number of facility elements from 16 for O_2/H_2 to 28 for FLOX/CH_4.

The number of operational tasks was determined by counting the functional blocks shown in Fig. 46 from beginning of off-pad operations until lift off, but not including abort or recycle. This operational approach assumes vehicle propellant loading on the pad from propellant storage facilities at the pad, and assumes that FLOX is mixed at the pad. The number of operations might be reduced in some cases by loading the vehicle direct from the propellant transporter without going through a pad storage facility. FLOX might be pre-mixed by the supplier, eliminating the need for mixing facilities and operations at the pad and reducing operational tasks to 31 and facility elements to 19.

The number of facility elements was determined by counting those items on Figs. 52 through 58 which were judged to contribute an element of complexity. The items included are listed for each propellant as follows:

- N_2O_4 (Fig. 52) – storage tank, transfer lines, fill rate control valve, facility flow control, propellant level instrucmentation

Table 34

COMPLEXITY COMPARISON

Propellant	N_2O_4/A-50	O_2/H_2	F_2/NH_3	OF_2/B_2H_6	F_2/H_2	FLOX/CH_4
Total No. of Operational Tasks	21	30	32	36	36	38
Total No. of Facility Elements	10	16	18	23	22	28

Notes:

1. FLOX mixed at the pad
2. Vehicle tanks loaded at the pad from pad propellant storage facilities

- A-50 (Fig. 52) – Same elements as N_2O_4
- O_2 (Fig. 53) – storage tank, vacuum jacket on storage tank, transfer lines, fill rate control valve, facility flow control, propellant level instrumentation
- CH_4 (Fig. 55) – Same as O_2
- NH_3 (Fig. 55) – Same as O_2
- F_2 (Fig. 56) – storage tank, vacuum insulation on storage tank, LN_2 jacket on storage tank, transfer lines, LN_2 jacket on transfer lines, fill rate control valve, facility flow control, propellant level instrumentation, passivation system, drying system moisture sensing, vent gas disposal system.
- OF_2 (Fig. 56) – Same as F_2
- FLOX (Fig. 57) – O_2 storage tank, F_2 storage tank, FLOX storage tank, vacuum jacket on O_2 storage tank, vacuum jacket on F_2 storage tank, vacuum jacket on FLOX storage tank, LN_2 jacket of F_2 tank, LN_2 jacket on FLOX storage tank, O_2 transfer lines, F_2 transfer lines, LN_2 jacket on FLOX transfer lines, LN_2 jacket on F_2 transfer lines, LN_2 jacket on FLOX transfer lines, fill rate control valve, propellant level instrumentation, facility flow control, moisture sensing, passivation system, drying system, FLOX composition sensing, FLOX composition control, vent gas disposal.
- H_2 (Fig. 54) – storage tank, vacuum jacket on storage tank, multilayer insulation on storage tank, transfer lines, vacuum jacket on transfer lines, multilayer insulation on transfer lines, fill rate control valve, facility flow control, propellant level instrumentation, condensible/reactant sensing.
- B_2H_6 (Fig. 58) – storage tank, GN_2 jacket on storage tank, transfer lines, GN_2 jacket on transfer lines, fill rate control valve, propellant level instrumentation, facility flow control, drying system, moisture sensing, air sensing, remote located burn vent stack.

3.5 REFERENCES

The following references were used in the analysis of ground operational requirements and problems:

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